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Final Report for a Robotic Exploration Mission to Mars and Phobos

PROJECT AENEAS

Response to RFP Number ASE274L.0893

submitted to:

Dr. George Botbyl
The University of Texas at Austin
Department of Aerospace Engineering
and Engineering Mechanics
Austin, Texas 78712

submitted by:

Argos Space Endeavours

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Argos Space Endeavours
Project Aeneas Design Team

Fall 1993

Chief Executive Officer
Justin H. Kerr

Chief Engineer
Erin Defossé

Chief Administrator
Quang Ho

Engineers
Ernisto Barriga
Grant Davis
Steve McCourt
Matt Smith

Aeneas Project

Preliminary Design of a Robotic Exploration Mission to Mars and Phobos

Approved: ______________________

Justin H. Kerr

CEO, Argos Space Endeavours

Approved: ______________________

Erin Defossé

Chief Engineer, Argos Space Endeavours

Approved: ______________________

Quang Ho

Administrative Officer, Argos Space Endeavours

Argos Space Endeavours

University of Texas at Austin

Department of Aerospace Engineering and Engineering Mechanics

November 1993

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Introduction

During the fall 1993 semester, Argos Space Endeavours (ASE), in cooperation with the University Space Research Association (USRA), NASA - Johnson Space Center (JSC), and the University of Texas Department of Aerospace Engineering completed a preliminary design of Project Aeneas, a robotic exploration mission to both Mars and Phobos.

The beginning of this final report discusses the project objectives and provides a summary of the Aeneas mission. Subsequent sections provide detailed explanations of the various elements of Project Aeneas developed by ASE including science, spacecraft, probes, and orbits and trajectories. The report concludes by describing the management procedures and project costs.

Project Objectives

Three main objectives drive the design of Project Aeneas. First, the mission must provide data to aid in determining a site on Mars suitable for a piloted landing. ASE proposes to achieve this objective through remote sensing of Mars, followed by the deployment of probes to the Martian surface, verifying the remote sensing data. To further aid the site selection process, Project Aeneas includes an investigation of the surface geology and weather patterns on Mars through the use of additional surface probes and penetrators.

The second objective given to ASE includes proving the concept of producing fuel on Mars from primarily indigenous materials. ASE addresses this concept, termed In-Situ Resource Utilization (ISRU), by collecting carbon-dioxide from the Martian atmosphere, adding hydrogen brought from Earth, and, after heating, producing methane through a process known as the Sabatier reaction.

A third and final objective of Project Aeneas is the analysis of the composition of the Martian moon Phobos. Project Aeneas design includes a penetrator device, targeted at the crater Stickney on Phobos, to return data on the chemical and geological properties of the Martian moon.

Mission Summary

The entire Aeneas mission comprises three spacecraft, launched via two Soviet Proton rockets. The first launch will deliver one Mars orbiter and one Phobos probe delivery spacecraft. The launch of the second Proton will transport a second orbiter to Mars. Each of the orbiters contain remote sensing instruments, surface probes and penetrators, as well as an ISRU device. The Phobos probe delivery spacecraft carries the Phobos probe as well as additional remote sensing apparatus.

Science Element

The three main objectives of the Aeneas mission drive the science element of the project and are reiterated below:

- Return data to Earth to aid in the determination of future piloted landing sites,
- Prove the ISRU fuel production concept, and
- Provide data to determine the composition of Phobos.

The science element suggests the following strategies to complete these objectives:

- Remote sensing of Mars to compliment existing data,
- Deployment of probes to the Martian surface,
- Deployment of the ISRU test facility to Mars,
- Remote sensing of Phobos, and
- Deployment of probes to Phobos.

Three instruments compose the Mars remote sensing strategy of the science element: a Gamma Ray Spectrometer (GRS), a High Resolution Camera (HRC), and a Thermal Emission Spectrometer (TES). The GRS returns low resolution data on the elemental composition of the targeted surface. The HRC produces detailed images of possible landing sites, and the TES provides information on the temperature and dust loading of the Martian atmosphere. ASE selected these instruments based on the type of data returned by the instrument and by the mass, power, cost, and volume constraints on the Aeneas mission.

ASE selected seven different types of instruments for the probes of Project Aeneas. These instruments include a Seismicity Network (SEIS), an Atmospheric Structure Instrument (ASI), a Mossbauer Spectrometer (MBS), an Alpha Proton X-Ray Spectrometer (APXS), a Thermal Analysis/Evolved Gas Analyzer (TA/EGA), a Surface Imager (SI), and a Meteorology Network (MET). The SEIS returns data about the seismic nature of the Martian geology, while the ASI provides data on the altitude varying properties of the atmosphere. The MBS and APXS analyze respectively the iron compounds and elemental composition of a Martian soil sample. The actual compounds present in the soil are revealed by the TA/EGA, and the SI produces stereoscopic images of the surface. Lastly, the MET relays meteorological information such as wind speed and direction, temperature, particulate density, and humidity.

The Aeneas ISRU concept proposes to make methane from carbon-dioxide combined with onboard hydrogen, thus meeting the second objective of the element. Due to mass and power constraints, ASE proposes an ISRU design which collects, compresses, and heats a Martian atmospheric sample using the kinetic energy of the probe as it descends from orbit. This atmospheric sample, composed of

97% carbon-dioxide, once mixed with hydrogen and heated, produces methane and water through the Sabatier reaction shown below

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O$$

Sensors in the Sabatier reactor will then detect the presence of methane, proving the concept of fuel production on Mars.

To fulfill the last science objective, the ASE science element uses a similar remote sensing and probes approach. Remote sensing of Phobos will be accomplished via a GRS unit, similar to the GRS on the Mars orbiter. Project Aeneas also includes the deployment of two probes to the surface of Phobos. One probe will target the crater Stickney on Phobos, providing the probe with increased access to the interior of Phobos. The ejecta found inside and near Stickney may also yield important information about the composition and geology of Phobos. The second probe adds redundancy to the Phobos mission and incorporates the flexibility to analyze an additional site.

Spacecraft Element

In order to provide redundancy and avoid a single catastrophic failure of Project Aeneas, ASE chose to launch three separate spacecraft, named Mars-Silva 1, 2, and 3, each containing different instrument packages. To simplify the design, the Common Spacecraft Bus (CSB) provides the structural base of each of the Mars-Silva spacecraft. Figure 1 is a drawing of the Mars-Silva spacecraft. The mass of each spacecraft is approximately 1100 kg, and all of the Mars-Silva units comprise an orbiter, a probe deployment module, and an R-40B engine. ASE estimates each spacecraft will cost under the $150 million budget for "discovery" class missions. ASE estimates the three spacecraft will cost approximately $400 million total.

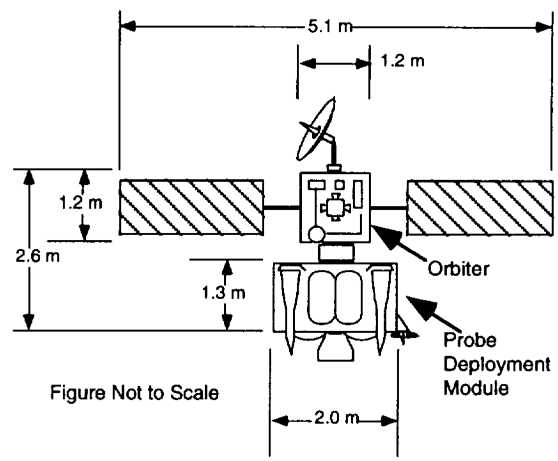

Figure 1 Drawing of the Mars-Silva Spacecraft

Even though each spacecraft has approximately equal mass, the probe configurations of each Mars-Silva vehicle differ. Mars-Silva 1 will deliver five science penetrators, one ISRU probe, and one canister lander containing two micro-rovers. The Mars-Silva 2 vehicle carries two Comet Rendezvous Asteroid Flyby (CRAF) type penetrators for deployment to Phobos. Lastly, the Mars-Silva 3 spacecraft holds three science penetrators, one additional ISRU probe, and two canisters each delivering two micro-rovers.

Even though Project Aeneas calls for three spacecraft, only two launch vehicles will be necessary. A single Proton rocket will launch both Mars-Silva 1 and 2 simultaneously, yielding a total injected mass of approximately 2055 kg. An additional Proton will carry the 1150 kg Mars-Silva 3 vehicle. A D1e upper stage engine provides a C3L injected mass capability of 5400 kg for each launch vehicle. ASE estimates a total mission launch cost of $80 million.

The guidance, navigation, and control of the Mars-Silva spacecraft includes three-axis control mechanisms and guidance mechanisms. Specifically, momentum wheels will provide three-axis control, and thrusters will function as an outlet for momentum dumping. Guidance is provided by the CSB. The CSB contains horizon sensors for simple guidance measurements, gyroscopes for measurements requiring high-accuracy, and a star tracker to calibrate the gyroscopes.

The three Mars-Silva vehicles receive electrical power through two means: silicon solar cells, and NiH_2 batteries. The deployable solar cells provide 190 W of power, enough power for all spacecraft operations. The NiH_2 batteries produce only 51 W of power. The batteries provide power primarily during blackout periods of the spacecraft. Approximately 10,000 light/dark cycles are expected during the mission. During these periods, the batteries provide enough power to operate the attitude control system, computer, and either the communication system or one scientific instrument.

ASE determined that the Rockwell RI 1750A/B computer would satisfy the data management needs of the Mars-Silva spacecraft. This computer provides 1750 instruction set architecture, a 16-bit processor, 1.8 Mips throughput, and 3.9 megabytes of storage. The Rockwell computer requires 7 W of power and has a mass of 2.5 kg.

Probes Element

ASE defined three requirements for the probes to achieve a successful exploration of Mars:

- Provide long duration science stations,
- Obtain readings from diverse locations on Mars, and
- Execute seismic, meteorological, and geoscience experiments.

To fulfill these requirements, ASE identified four types of probes: Mars penetrators, Phobos penetrators (using CRAF technology), landers containing micro-rovers, and the ISRU probe.

Mars penetrators form the backbone of the Aeneas probe fleet. Figure 2 shows a typical Martian penetrator. Penetrators enter the atmosphere from orbit and deploy drag bodies to slow the probe to a safe impact velocity. On impact with the surface, the penetrator submerges approximately two-thirds of its length into the surface of the planet. Penetration of the surface allows for the collection of deep soil samples for analysis, and gives the probe a firm base for seismic measurements. Communications are relayed back to Earth via the orbiting Mars-Silva spacecraft.

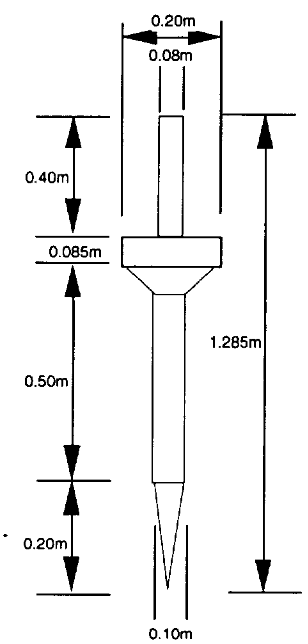

Figure 2 Drawing of Martian Penetrator

Due to the absence of an atmosphere and a weak gravitational field at Phobos, the Aeneas Phobos probe utilizes CRAF-type, proximity operations techniques to navigate around and penetrate Phobos. After penetration, the mission of the Phobos probe is similar to that of the Mars penetrator; the Phobos probe analyzes samples from beneath the surface of Phobos for their chemical and geological characteristics. Mars-Silva 2 will deliver the Phobos probe and relay probe data back to Earth.

The Mars lander and micro-rover combination constitute the Martian surface operations for Project Aeneas. The primary function of the lander is delivery of the micro-rovers to the surface. The lander also relays communications from the micro-rovers to the Mars-Silva orbiter. Micro-rovers carry either a single APXS or MBS instrument. A micro-rover can travel approximately 20 meters per day, analyzing samples along its path.

Orbits and Trajectory Element

ASE adopted five design strategies in establishing the orbit and trajectory for the Aeneas mission:

- To design a "typical" transfer trajectory to Mars,
- Use Hohmann transfer data to carry out preliminary mission design,
- To size the spacecraft using Hohmann data,
- Use Lambert targeting to refine the initial trajectory calculations, and
- To identify spacecraft and launch system requirements based on an optimized trajectory.

In the design of the Aeneas trajectory, ASE first calculated a Hohmann transfer trajectory, producing a time of flight of 258 days and a C3 of 8.6 km^2/s^2. Next, ASE optimized the Hohmann trajectory using Lambert targeting techniques. This included identifying launch opportunities and C3 requirements using "pork chop" plots, provided by the Jet Propulsion Laboratory (JPL). The orbits and trajectory element then optimized the trajectory for minimum launch C3, minimum arrival C3, and launch and arrival dates. Lastly, ASE identified booster and upper-stage combinations which satisfy the launch C3 and spacecraft mass requirements. The Lambert trajectory optimized for minimum launch C3 gave a time of flight of 202 days, and a launch C3 of 8.8 km^2/s^2. Figure 3 shows a plot of the Earth-Mars Lambert trajectory.

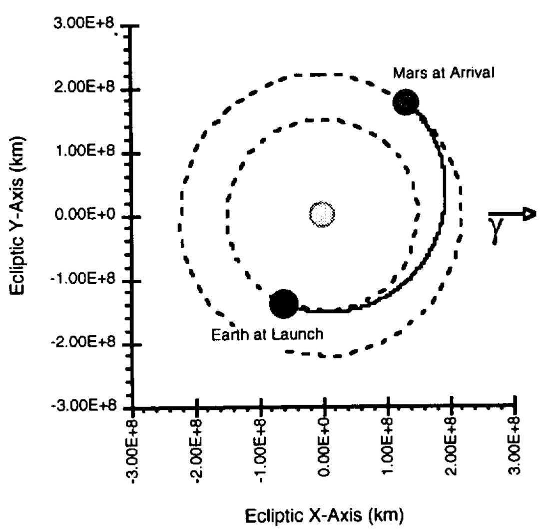

Figure 3 Plot of Earth-Mars Lambert Trajectory

The general scheme of orbit insertion follows three distinct paths. First, Mars-Silva 1 and 2 separate during the transfer trajectory. Next, Mars-Silva 1 inserts into a 60° inclination orbit about Mars. Mars-Silva 2 diverges into a near equatorial orbit, closer to the orbital plane of Phobos. When Mars-Silva 3 arrives, the spacecraft enters a 60° inclined orbit, similar to Mars-Silva 1.

The orbits for Mars-Silva 1 and 3 have a semi-major axis of 3880 km, an altitude of 483 km, an eccentricity of nearly zero, and an inclination of approximately 60°. This orbit gives coverage of ±60° latitude and requires 12 revolutions to obtain a near-repeat ground track. The orbiter should be able to completely map the surface in approximately one year.

Recommendations

The following list of recommendations are areas of Project Aeneas which require further development.

- Develop the ISRU vehicle in detail (possibly a project that should be handled by ASE 363Q)

- Develop the Penetrator structural design (possibly a project that should be handled by ASE 363Q)

- Carry out a more in-depth analysis of the trajectory issues. In particular the targeting of the spacecraft into Mars orbits and the final form of the Mars orbits themselves.

- Carry out a more in-depth analysis and design on the spacecraft and its subsystems. The work carried out by ASE is preliminary and is only intended to provide an overall spacecraft design which would be suitable for a mission like Project Aeneas.

- Investigate the targeting issues involved in delivering probes to the surface of Mars and Phobos. In particular, develop a model for the thermal environment that the probes will encounter upon entering the Martian atmosphere. Also, develop guidance and control systems which will ensure that the probes are delivered accurately.

- Carry out a detailed study on how the orbiters will map the surface of Mars in preparation for the deployment of probes. Generate ground tracks and figure out how (in terms of orbit design) to maximize the coverage of interested locations on the surface.

- Consider adding studies of micro meteoroid impacts on the Martian surface and radiation levels during the cruise phase to Mars.

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1.0 Introduction

This final report is written in response to RFP number ASE274L.0893 to design a robotic exploration to Mars and Phobos. This report begins with a discussion of the missions's background and objectives, and continues with detailed explanations of the various elements of Project Aeneas, including science, spacecraft, probes, and orbits and trajectories. In addition, a description of Argos Space Endeavours' management procedures and the overall project costs are presented. Finally, a list of recommendations for future design activity are included.

1.1 Mission Background

Renewed interest in both the exploration and settlement of space has brought an increase in the development of robotic exploration missions. These missions are designed to pave the way for human missions of the future, and their primary objectives include the search for potential landing sites. The selection of these sites depends upon such criteria as the ease of landing and the accessibility to sites of scientific interest.

Project Aeneas is a response to a request for the design of a robotic exploration mission to both Mars and Phobos. The primary goal of this project is the determination of suitable landing sites for the future, but it also includes several other objectives of significance. The first is based upon the necessity of a continuous fuel supply at any permanent station and the near certainty that this will require production from local materials. This concept of local production is termed In Situ Resource Utilization (ISRU) and the ability to produce fuel from indigenous Martian resources is the technology experiment of Project Aeneas. Another objective of this mission is the deployment of a probe to the Martian moon Phobos for determining its elemental composition.

1.2 Mission Objectives

The objectives of Project Aeneas as stated in the RFP are as follows:

- To determine an ideal landing site on Mars for future manned missions
- To explore and perform scientific experiments on the Martian surface.
- To launch a probe(s) to Phobos to determine its elemental composition.
- To relay the scientific/exploration data and images back to Earth.
- To design a proof-of-concept fuel/propellant production facility for deployment to the Martian surface.

1.3 Mission Specifications

The following are mission specifications as detailed in the RFP.

- Fully robotic mission.
- Focus primarily on the Mars end of the mission.
- Must include how to get to Low Mars Orbit.
- Lander should be a common lander design for exploration and sized appropriately for this type of mission.
- No large cargo lander.

- No return to Earth capability
- Include communications, data transmission to return scientific/exploration information and images to Earth.
- Include a small robotic proof-of-concept fuel production facility.
- Scientific and exploration rovers, probes, or other packages should be considered.

1.4 Additional Mission Requirements

In addition to the requirements established for Project Aeneas in the RFP, Argos Space Endeavours imposed additional requirements that affected the project's design. These additional requirements were added to meet the current political and economic environment. NASA is interested in smaller, cheaper missions which are designed to return at least limited data upon subsystem failure. The first constraint was the decision to include redundancy in the mission design. This decision was based on the recent Mars Observer loss. By taking a multiple vehicle and multiple launch approach, Argos Space Endeavours will create a mission with a low risk of total mission failure. This level of redundancy was constrained by the second internal decision, that each spacecraft be considered "discovery" class. This requirement means that each spacecraft must cost under $150 million, with the entire project capped at $500 million. In addition, the mission would have to utilize existing technology and be of limited scope. Each of these constraints were incorporated in the design of Project Aeneas.

2.0 Science Element

The science element is responsible for determining the manner in which the primary mission objectives will be fulfilled. These goals include the determination of:

- an "ideal" landing site for future human missions to Mars,
- the elemental composition of Phobos, and
- the proof of concept for fuel/propellant production using Martian resources.

The science element relays its requirements to the other elements which, in turn, develop the engineering aspects of the mission.

2.1 Science Element Strategies

In order to complete its objectives, the science element has developed the following strategies:

- remote sensing of Mars;
- deployment of probes to Mars, including the ISRU (in situ resource utilization) fuel test facility;
- remote sensing of Phobos, and
- deployment of a probe to the crater Stickney on Phobos.

2.2 Knowledge Required for Human Missions

The primary goal of Project Aeneas is to aid researchers in determining an "ideal" landing spot for future human missions to Mars. Before astronauts can be sent to the planet, conditions must be known which will ensure their safety from arrival to departure as well as their ability to travel to places where they can be used effectively.

By more fully understanding Martian science issues, the value of landing humans on the planet can be substantially increased. To this end, one aim of Project Aeneas is to compliment and refine the existing data for Mars. This goal will in turn help to define the science that humans will perform.

2.2.1 Physical Properties and Chemistry of the Surface Materials

Imaging from the two Viking landing sites show areas that have large concentrations of rocky debris to the degree that it could endanger landing and hinder movement on the surface. Although it is unknown how typical these sites are of the Martian surface, infrared observations tend to suggest that they are rougher than average. Modeling of this data, however, cannot distinguish between bedrock and surface particles; thus, a large rock covered with several centimeters of regolith may not be detectable. [S 1;21] Current orbital imaging, obtained from the Viking mission , has near global coverage at 200 m/pixel resolution with small sections ranging up to 8 m/pixel [S 1;29]. This is too low of a resolution to see surface rocks. To determine and judge potential hazards such as rocks, steep slopes, and crevasses, imaging must be obtained at sub-meter per pixel resolution either from orbit or from the surface. Previous studies have determined that the detection of objects 1 m in size is a reasonable goal and simulations have shown that this would require an imaging system with resolution in the range of 20-30 cm/pixel. [S 1;36]

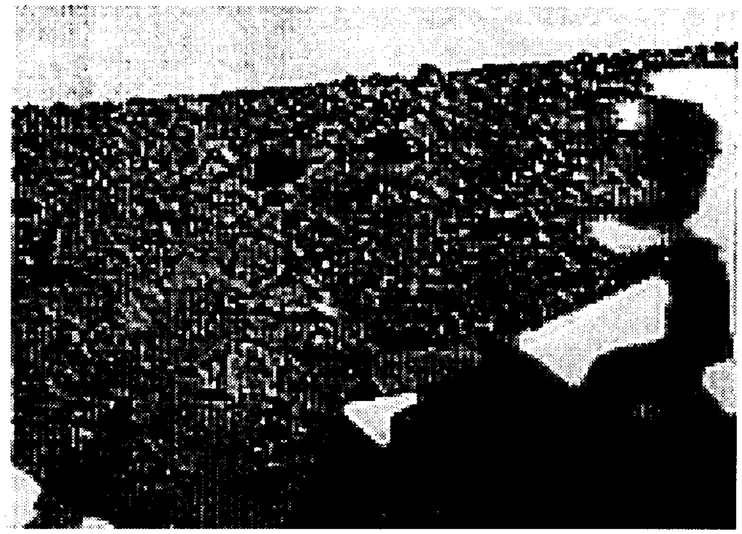

Figure 2.2.1- 1. Scene from the Viking 2 Lander

Because of its potential for radiation shielding, a protective layer of regolith could be desirable at a human base. Images from the Viking 1 lander seem to show that the regolith at that site is thin; however, we have no knowledge of the thickness elsewhere. Infrared measurements indicate that regolith is present on nearly all of the Martian surface, but this data refers to only the upper centimeters and gives no indication of the absolute thickness. [S 1;21] Although it is desirable to locate a human base near the useful regolith, a thick layer of regolith could cause some concern at a landing site [S 2;20]. Data from Viking 1 suggests that the regolith has the consistency of flour, and one of the lander's footpads sank deeply into the surface. If the regolith was too thick it could pose a hazard to landing as well as rover and human mobility. Acquisition of this data will require in situ measurements, such as those provided by penetrators, geophysical sounding, and/or roving vehicles. [S 1;21]

Unlike lunar regolith, that of Mars is almost certainly not inert and contains reactive chemical species such as oxidants. [S 1;22] Knowledge of its chemistry is required to determine its possible reaction with different materials and humans. In addition, its potential to serve as a source for construction materials can be better determined with detailed elemental and chemical analyses. While the Viking landers had biological experiments, a lander with more diverse instruments and experimental capabilities would be able to provide these necessary analyses.

2.2.2 Atmospheric Properties

For human bases, the characteristics of the Martian atmosphere that are of primary concern are wind speed and the dust loading associated with it. Structures as well as roving vehicles would almost certainly be affected by the dust. Peak winds are associated with the Martian dust storm activity which varies widely from year to year in location, size, and timing. [S 1;22] Because we do not understand the reason for these variations, our ability to predict dust storms is limited. Determination of these conditions at each potential landing site will require in situ measurements. In this manner, we will be better able to predict the conditions at these sites.

2.3 Orbiter Instruments

The following remote sensing instruments were selected for their ability to complete the scientific goals of this project. For further descriptions of these instruments see Appendix S-A.

The primary goal of the gamma-ray spectrometer (GRS) will be to measure the elemental composition of the Martian surface with its spatial resolution of 360 kilometers. [S 3;491] In addition to investigations of Mars, the GRS can address some of the problems in solar physics and astrophysics. These problems include the high energy processes in solar flares and gamma-ray bursts of stellar and nonstellar objects. [S 4;3]

The high-resolution camera (HRC) is intended to provide detailed surface characteristic data which will facilitate the selection of potential landing sites. In addition, it will enable scientists to monitor surface and atmospheric features over time and to systematically examine local areas at high resolutions (sub-meter per pixel). Ballistic Missile Defense Organization (BMDO) technology may be utilized for this instrument.

The thermal emission spectrometer (TES) will enable scientists to obtain a variety of information about the surface and the atmosphere of the planet. In particular, the TES will provide information about the variations of surface mineralogy [S 1;29] and determine the atmospheric profiles of temperature, pressure, water vapor (H_2O), and ozone (O_3). [S 4;24] The combination of the TES and GRS will better enable scientists to judge what types of volcanic and sedimentary rocks are present on the Martian surface which will be the primary basis for determining what materials to expect at potential landing sites.

2.4 Mars Probe Instruments

The following instruments, chosen for the Martian probes, were selected for their ability to return data required by the goals of Project Aeneas.

The Alpha/Proton/X-ray Spectrometer (APXS) instrument will be carried to the surface of Mars to determine the elemental composition of the soil and rocks in the vicinity of the lander. Used by placing the sensor head against a sample, the APXS will return the elemental composition of that subject for most major elements except hydrogen. [S 5;4] Due to the relatively small size and power requirements of the APXS, it is an ideal instrument to be carried on small rovers.

The Mössbauer spectrometer (MBS) will be deployed to the surface on either small rovers or the full-science lander depending on final mass and power requirements of the instrument (still under development). MBS is designed to determine the iron mineralogy of the soil by identifying individual phases of iron ore in the soil. MBS can therefore provide data on the chemical, not merely elemental, composition of the soil.

The meteorology package (MET) is a system of instruments designed to provide information on the atmospheric conditions at each landing site. These instruments will measure atmospheric pressure, wind velocity, humidity, and temperature over the course of a Martian year.

The atmospheric structure instrument (ASI) is a decent instrument which measures pressure and temperature during descent. In addition, an accelerometer placed near the center of gravity of the lander will measure peak accelerations during entry. [S 5;4] This instrument will help in creating a model of the Martian atmosphere for use in designing human landing systems.

The thermal analyzer/evolved gas analyzer (TA/EGA) works by heating up a soil sample and analyzing the gases that are produced. This analysis will provide more detail about the compounds present at each of the landing sites. The gas analyzer measures water content, carbon, nitrogen, oxygen, organic content, and oxidants. [S 5;4]

A three-axis seismometer (SEIS) will measure ground motions during seismic events. By placing a number of these instruments over a large area of the Martian surface, information can be gained about the internal structure of the planet. [S 1;31]

2.5 In-Situ Resource Utilization (ISRU) Fuel Production Facility

The RFP requires Project Aeneas to design a proof-of-concept fuel production facility which will be dispatched to the Martian surface. Argos Space Endeavours proposes to send an in-situ resource utilization (ISRU) lander. In other words, the experiment will prove the concept of in-situ fuel production for future missions. To fulfill this objective, the Aeneas team decided to produce methane, a potential fuel for facilities on Mars [S 6;1], from the carbon-dioxide present in the Martian atmosphere. A common chemical process, the Sabatier reaction, efficiently produces methane and oxygen from carbon-dioxide and hydrogen [S 7;272]. The Martian atmosphere consists of approximately 95% carbon-dioxide [S 2;17], and this resource is readily available everywhere on the surface of the planet. Thus, Argos Space Endeavours (ASE) decided to utilize Martian atmospheric carbon-dioxide in a Sabatier reaction with hydrogen to produce methane as a fuel and oxygen as an oxidizer. Due to the scarcity of hydrogen on Mars, hydrogen would need to be brought with the Aeneas ISRU spacecraft.

2.5.1 ISRU Mission Profile

One of the challenges faced by ASE in using the Sabatier reaction is the difficulty in compressing and heating the carbon-dioxide to suitable pressures and temperatures. Delivering compressors and large heaters to the surface of Mars would exceed the mass and power constraints of the Aeneas mission. Thus, ASE decided to collect, compress, and heat a sample of the Martian carbon-dioxide during the entry of the ISRU device into the atmosphere. Figure 2.5.1-1 contains a mission profile summarizing the Aeneas ISRU mission.

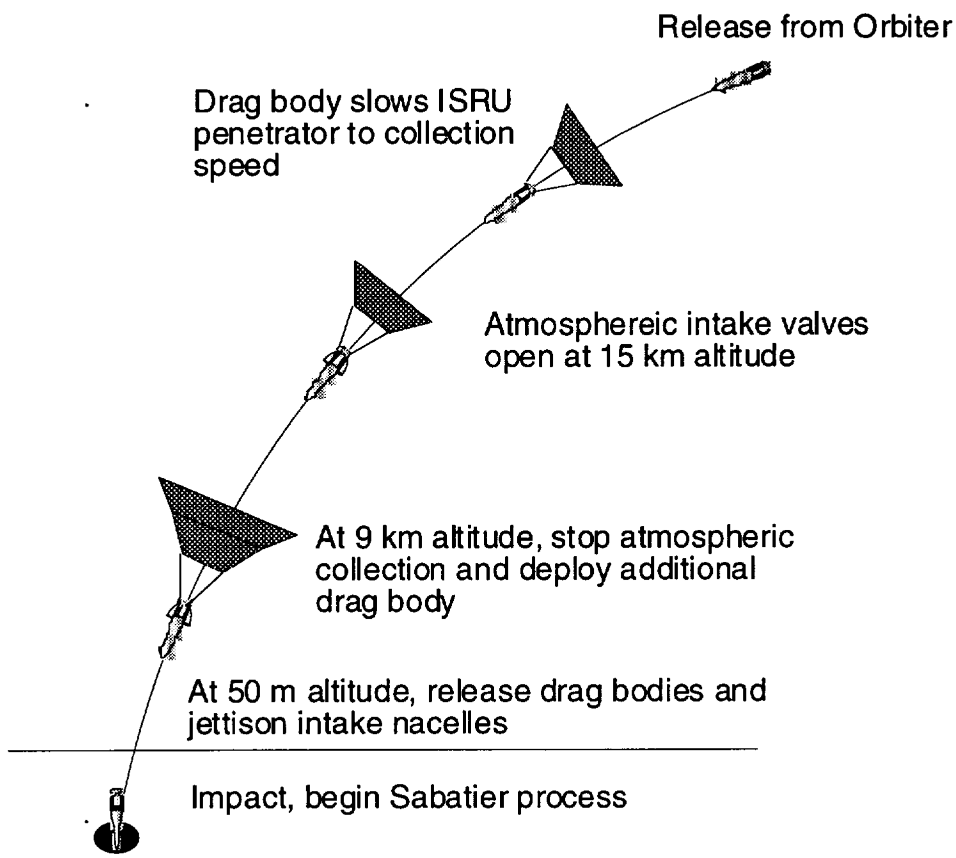

Figure 2.5.1-1: Aeneas ISRU Mission Profile

2.5.2 ISRU Chemistry

When fully implemented, the Sabatier reaction consists of three steps [S6;1]. First, the main reaction, shown in equation [1], combines carbon-dioxide and hydrogen.

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O \qquad [1]$$

After heating the gaseous mixture, methane and water are produced. A second electrolysis reaction, shown in equation [2], separates the water into hydrogen and oxygen.

$$2H_2O \rightarrow 2H_2 + O_2 \qquad [2]$$

The hydrogen produced in the electrolysis reaction is then recycled into the main reaction. Combining equations [1] and [2] creates a net reaction, as shown in equation [3], producing methane (a fuel) and oxygen (an oxidizer).

$$CO_2 + 2H_2 \rightarrow CH_4 + O_2 \qquad [3]$$

In order to simplify the hardware necessary to carry out the chemical reactions, ASE chose to perform only the reaction given in equation [1] above. In this case, the methane produced would be detected by sensors in the reaction vessel and no hydrogen would be recycled However, in an actual full scale fuel production facility, all three reactions given above would need to be implemented to conserve hydrogen.

To produce one kilogram of methane, the ISRU device needs to collect 2.74 kilograms of carbon-dioxide as shown in equation [4] below.

$$1000gCH_4 \cdot \frac{1molCH_4}{16.04gCH_4} \cdot \frac{1molCO_2}{1molCH_4} \cdot \frac{44.01gCO_2}{1molCO_2} = 2740gCO_2$$

2.5.3 ISRU Atmospheric Collection

To meet mass and power constraints, ASE decided to collect, compress, and heat an atmospheric sample to Sabatier reaction conditions using the energy of the descending ISRU device. The mission plan specifies opening the collection intakes of the ISRU device at 15 km altitude and closing the intakes at 9 km altitude. A collection cutoff point at 9 km altitude was chosen in order to allow enough altitude to slow the ISRU device to a safe landing velocity.

The starting collection altitude was determined by calculating the altitude at which a 60% efficient cylindrical intake would collect enough carbon-dioxide to produce one kilogram of methane. ASE used the COSPAR Martian atmospheric model, provided by the Jet Propulsion Laboratory (JPL), to determine the properties of the atmosphere. A copy of the FORTRAN code used to determine the collection altitude, called cylinder.f, is given in Appendix S-B. By iterating over 10 meter increments, the cylinder.f program predicted an initial collection altitude of 15 km for the ISRU device to collect sufficient carbon-dioxide for the proof of concept mission.

2.5.4 ISRU Mission Configurations

The ISRU mission comprises two distinct configurations of the probe: the flight configuration and the surface configuration. Figure 2.5.4-1 shows the configuration of the ISRU unit during flight including a view of the internal reaction vessel.

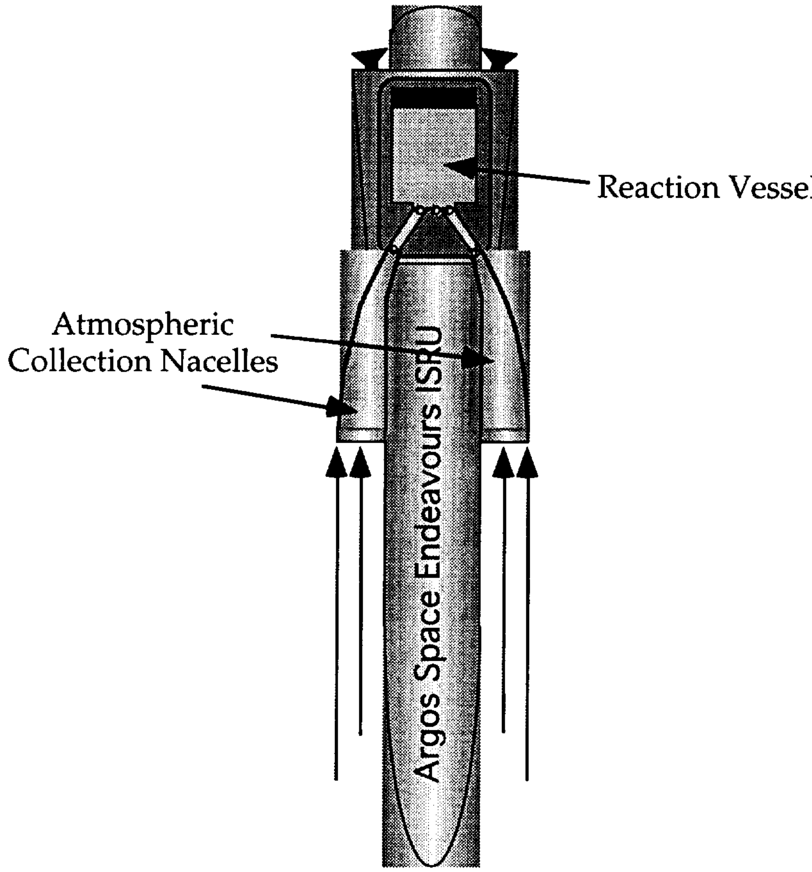

Figure 2.5.4-1: ISRU Flight Configuration

In the flight configuration, Martian atmosphere flows through the atmospheric collection nacelles and into the reaction vessel.

On the surface, the ISRU probe assumes a configuration to carry out the Sabatier reaction, as shown in Figure 2.5.4-2 below.

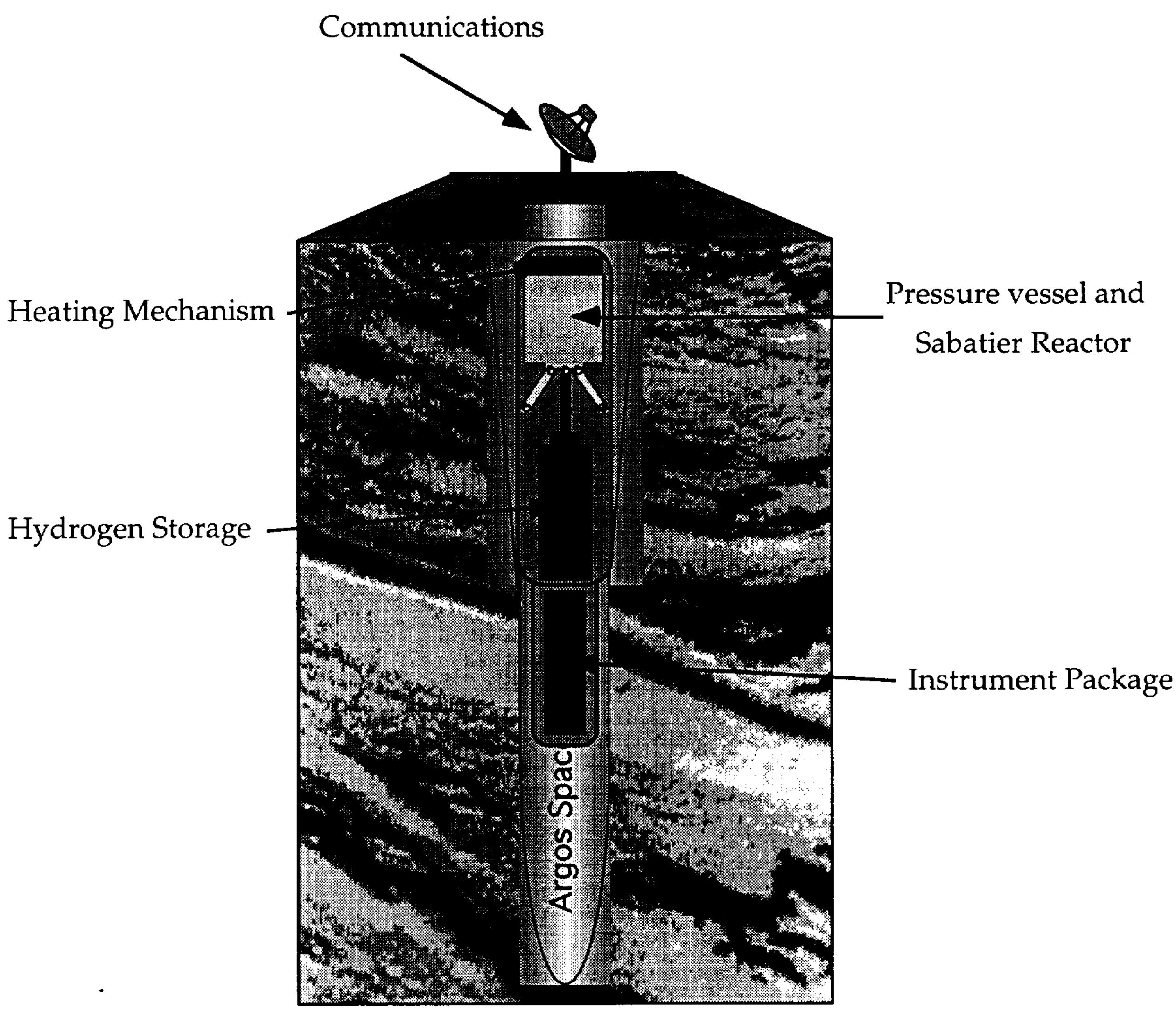

Figure 2.5.4-2: ISRU Surface Configuration

The first step in surface operations is the addition of hydrogen to the Sabatier reactor. Next, the heating mechanism adds thermal energy to the reactor which initiates the Sabatier reaction. As the reaction proceeds, sensors in the reaction vessel detect the presence of methane and relay this information to the instrument package. Lastly, the readings of the instruments are relayed back to the orbiter via the ISRU communications equipment.

2.6 Phobos Probe Instruments

The theory that Phobos, the largest moon of Mars, is composed primarily of carbonaceous chondrites has fallen into debate. Due to the fact that Stickney is the largest crater on this moon, Argos Space Endeavours believes that a probe landing at this site will have better access to the interior of Phobos. This positioning in turn will aid scientists in determining the composition of the moon.

An APXS instrument (see section 2.4 for details) will be carried to the surface of Phobos on a penetrator type probe to conduct spectroscopy studies intended to determine the elemental composition of the soil. The survivability of APXS instruments to shock is still being examined. However, the Russian Mars '96 instrument expects to carry an APXS on a penetrator for use on Phobos. This mission will determine if sending such an instrument on a probe is feasible.

2.7 Selection of Instruments

Although a number of instruments were initially considered for inclusion in Project Aeneas, budget considerations limited the type and number that could actually be selected. Such considerations were cost, mass, power requirements, and size. The primary factor in determining which instruments to carry, however, was the ability of each of the potential instruments to provide data that would lead to the completion of the scientific objectives. The remote sensing instrument of highest priority was thus determined to be the high resolution camera (HRC) because of its ability to provide sub meter per pixel images of the potential landing sites which is a necessary precursor to a human mission.

The final two remote sensing instruments, the gamma-ray spectrometer (GRS) and thermal emission spectrometer (TES) are very close in their importance to Project Aeneas. Due to the GRS's ability to give researchers a global picture of the elemental distribution of the planet, it was ranked slightly higher than the TES.

3.0 Spacecraft Element

Based on the scientific strategies outlined in section 2.0, Argos Space Endeavours has determined that the Mars-Silva spacecraft design must achieve the following tasks:

- provide communication with Earth and Mars/Phobos probes,
- conduct remote sensing of the Martian surface,
- transport orbiting platforms/probes to Martian orbit, and
- deploy probes to the surface of Mars and Phobos.

Each spacecraft is divided into two sections, the orbiter and the Probe Deployment Module (PDM). The orbiter will provide communication and remote sensing capabilities. The PDM, on the other hand, will support the probes during the journey to Mars and deploy the probes to the surface of Mars and Phobos. The PDM is required in order to prevent probe deployment from interfering with orbiter functions.

3.1 Common Spacecraft Bus

For the orbiter, we will be using a Common Spacecraft Bus (CSB) designed by B. N. Agrawal at the Naval Postgraduate School in Monterey, CA [Sp 1;1]. The CSB was designed to perform different missions with a common vehicle design. After deciding which types of subsystems to use for the orbiter, the modified CSB was found to meet all the requirements for completion of the mission.

The CSB uses six Rocket Research Model MR103C hydrazine thrusters and a propellant tank with a 20 kg capacity. Each thruster produces 0.89 Newtons of force [Sp 1;8], or 0.45 N•m of torque in the yaw direction.

The CSB uses a horizon sensor and gyroscopes for simple attitude determination, three gyroscopes for accurate short-term measurements, and a star sensor to periodically calibrate the gyroscopes [Sp 1;7].

The thrusters and momentum wheels of the Common Spacecraft Bus [Sp 1;7] are sufficient for the attitude control required by the spacecraft (see section 3.3.4). The thrusters will be used for desaturation.

The CSB uses a silicon solar cell array and NiH_2 batteries for power. The CSB is regulated at 28V with a shunt regulator for operation under solar power and a boost regulator for battery operation. The CSB experiences a 43% lifetime degradation in solar panel performance due to exposure to the Van Allen belts, and 9% degradation for the mission where it is not exposed to the belts. These degradations occur over 3 years, compared to the Project Aeneas mission length of one year in Mars orbit.

3.2 Reasons for Using the Common Spacecraft Bus

The reasons for using the Common Spacecraft bus follow:

- The CSB has been designed for different missions; two sample missions are described in the CSB reference. Use of a multi-mission spacecraft bus reduces production costs and improves reliability.

- With some increases in structural integrity, the CSB meets our needs for the Project Aeneas orbiter. It has sufficient solar power, which was chosen in a trade study to be the orbiter's main source of power. Its attitude control system is more than adequate for our needs, and its guidance system will also meet our needs with some software modification.
- The CSB paper has data on certain spacecraft systems which would otherwise be difficult to obtain and verify. This helps to reduce development costs.

3.3 Romulus Class Orbiter Design

3.3.1 Orbiter Sizing

Figure 3.3.1-1 shows that the orbiter is less than 1.2 meters on each side. It was originally designed to fit inside the Pegasus payload shroud [Sp 2;8]. The solar panels extend to a total length of 5.1 m [Sp 2;3]. The solar panels have been determined to be large enough for the needs of the orbiter (see Table 3.3.5-4).

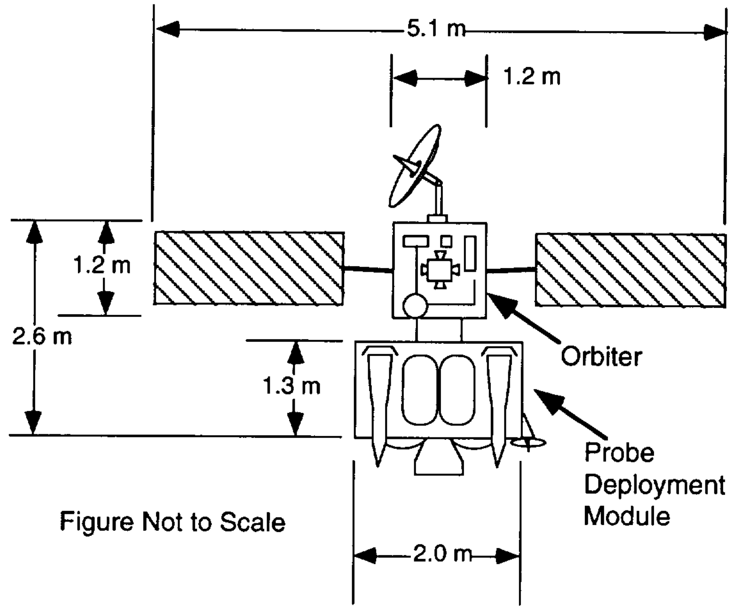

Figure 3.3.1-1: Drawing of Mars-Silva Spacecraft

3.3.2 Orbiter Mass

The orbiter mass was determined by a compilation of component masses from the Science element and the known component masses of the CSB. The instrumentation [Sp 4;7666], communications, and computation masses came from the Science element, while the attitude control, power, thrusters, thermal, electrical and mechanical integration, and propellant (increased to the capacity of the propellant tanks) masses are known for the CSB. The battery mass was determined from the night operation power budget (see section 3.4.5), and the CSB structure was scaled up from 25 kg to 37 kg to support the increased mass of the orbiter (250 kg as compared to 183.5 kg [Sp 1;4]). The orbiter mass budget is summarized in table 3.3.2-1.

Table 3.3.2-1: Orbiter Mass Budget

Component	Mass (kg)
Instrumentation	45
Attitude Control	25
Power	37
Thrusters	15
Structure	37
Thermal	5
Communications	20
Computation	3
Electrical and Mechanical Integration	7
Propellant/Pressurant	20
Battery	18
Margin	18
Total	**250**

3.3.4 Orbiter Guidance, Navigation, and Control

When choosing the method of controlling the Aeneas spacecraft, many options were considered. The spacecraft must take accurate reading of the Mars surface, maintain contact with Earth and the probes on the surface, and keep its solar panels continuously pointed directly towards the sun. Because of these accuracy requirements, three axis control will be used to control the orbiter's attitude [Sp 3;sec.2;4].

When an orbiter is maintaining three axis control, it must provide torques to turn the spacecraft at a given slew rate. It must also counteract the disturbance torques which may act on the satellite. The possible sources of torque on a spacecraft are gravity gradient, solar radiation, magnetic field, and aerodynamic forces [Sp 2;353].

In Earth orbit, some satellites use a gravity gradient to maintain the proper attitude with respect to the Earth. The gravity gradient is a torque caused by the difference in gravity forces from the end of the satellite which is closest to earth to the end which is the farthest away[Sp 3;sec.2;5]. Because of the lower gravity of Mars, the gravity gradient is a small effect and will not be used for attitude control although it will be calculated as a disturbance torque.

The most common methods of three axis control are thrusters, reaction and momentum wheels, and control moment gyros. Thrusters rotate the spacecraft by expelling mass along a moment arm[Sp 3;sec.2,33]. Momentum wheels apply a torque to the satellite by accelerating internal wheels in a direction opposite of the desired torque[Sp 3;sec.2,36]. Control moment gyros use one wheel rotating at a constant speed on a gimbal. Control moment gyros provide more torque, but have more mass and a greater power requirement[Sp 3;sec.2,37].

Momentum wheels or control moment gyros can become saturated when they reach their maximum angular velocity. The spacecraft then requires a moment in the opposite direction to "desaturate" the system [Sp 3;sec.2,37]. Because of saturation, thrusters are required even if momentum wheels or control moment gyros are used.

The type of attitude control system used depends on the spacecraft torque requirements. The required torque is equal to the worst-case disturbance torque on the orbiter plus the torque required to meet the turning needs of the orbiter.

The torque due to the gravity gradient for the PDM alone is

$$T_g = \frac{3\mu}{2R^3}\left|I_z - I_y\right|\sin(2\theta) = .000188\ \text{N}\bullet\text{m}, \quad [8]$$

where m is the gravity constant of Mars, R is the distance from the center of Mars to the orbiter, I_y and I_z are the moments of inertia of the satellite (see section 3.4.1), and q is the maximum deviation of the z axis from the local vertical. q is assumed to be 45 degrees, for the worst-case torque.

The torque due to solar radiation is

$$T_{sp} = \frac{I_s}{c} d_{pg} A(1+q)\cos i = 2.8{*}10^{-5}\ \text{N}\bullet\text{m}, \quad [9]$$

where I_s is the solar intensity (600 W/m^2 at Mars[Sp 3;sec.7,10]), c is the speed of light ($3.0{*}10^8$ m/s), d_{pg} is the distance from the center of pressure to the center of gravity (assuming a worst-case of 1.2 m), A is the exposed area of the spacecraft (6 m^2), q is the coefficient of reflectivity (0.9 for worst-case), and I is the angle of incidence (worst case of 0 degrees).

Because the magnetic field of Mars is a subject of great debate and generally considered to be small, a calculation of its disturbance torque is unavailable. Therefore, the magnetic field torque is considered to be negligible. Because Mars has almost no atmosphere at the altitude of the orbiter, air friction is ignored.

The orbiter requires a slew rate of .05 degrees per second. This is sufficient to keep the solar panels oriented normal to the sunlight for maximum power generation, and is typical for spacecraft which must maintain a local vertical with the planet.

The torque required for the given slew rate is

$$T_{sl} = \frac{4I_y q}{t^2} = .0028 \text{ N}\bullet\text{m}, \qquad [10]$$

where q is the slew angle and t is the given time (a slew of 30 degrees in 10 minutes is assumed)[Sp 2;357].

The total torque requirement is less than 0.01 N•m, the torque provided by the smallest momentum wheels[Sp 2;p.355]; therefore, we can use momentum wheels in our design. The attitude control system of the CSB will be sufficient for the needs of Project Aeneas. Note that thrusters will still be required for desaturation. Additional thrusters will be mounted on the PDM to provide redundancy and additional torque for a higher slew rate, if necessary.

3.3.5 Orbiter Power

The power requirements of the remote sensing instruments on board the orbiter are summarized in Table 3.3.5-1.

Table 3.3.5-1: Remote Sensing Power Requirements [Sp 4;7666]

Instrument	Power (average/peak) in W
Gamma Ray Spectrometer (GRS)	14.0/34.9
High Resolution Camera (HRC)	7.5/25.7
Thermal Emission Spectrometer (TES)	13.2/18.3

One Mars orbiter will carry a GRS and an HRC, while the other Mars orbiter will have a TES and an HRC. The orbiter with the Phobos probes will carry an HRC and a GRS. These instruments have an average power consumption of 21.5 W.

The orbiter has a total power requirement of 190 Watts when exposed to the sun, and a requirement of 51 Watts for night operation. The night power budget delineates the power requirements when the spacecraft is between the sun and Mars. During night operation, the orbiter must rely on intermittent thruster operation for attitude control and may not operate communications or the remote sensing instruments. Because the orbiter spends twice as much time in the sun as out of it and the batteries are assumed to charge with 40% efficiency, 125% of the night power requirement is needed during daylight operation to charge the batteries. The power budget of the orbiter is summarized in Table 3.3.5-2.

Table 3.3.5-2: Spacecraft Power Budget

Subsystem	**Power(W)**	**Night Power**
Instrumentation	30	0
Attitude Control	59	0
Thrusters	6	2
Thermal Control	10	10
Communication	23	0
Computation	7	7
Power (losses)	14	2
Total	149	21
Battery Charging	26	
Margin	15	
Power Required	**190**	

The different types of available power sources are: solar arrays, fuel cells, batteries, and Radioisotope Thermoelectric Generators (RTGs). Other types of power generation, such as solar dynamic systems and large nuclear plants, require too much mass and produce more power than needed[Sp 3;Sec.7].

Fuel cells and batteries have a very limited life span. For instance, the fuel cells on the Space Shuttle have a lifetime of 2000 hours, or approximately 40 days[Sp 3;sec.7,13]. In addition, fuel cells put out considerably more power (7000 W for the Shuttle, or 1000 W for the Gemini missions) than we require [Sp 3;sec.7,13]. Our mission has an estimated lifetime of 1 year or more in orbit around Mars. Therefore, we will not be using fuel cells. The orbiter's NiH_2 batteries will provide the needed power until the spacecraft reaches Mars.

A trade study was performed to determine whether solar panels or RTGs should be used to provide power to the orbiter. This required sizing of solar panels and RTGs for a given power output. A power output of 200 Watts was assumed for the purposes of the trade study.

Table 3.3.5-3 shows the parameters for preliminary sizing of a solar array with a power output of 200 Watts[Sp 3;sec.7,10].

Table 3.3.5-3: Solar Panel Sizing Characteristics

Material	Silicon
Solar Cell Efficiency (E)	11.5%
Power load at End-of-Life (EOL)	200 W
Packing Efficiency (E_p)	90%
Temperature Efficiency(E_t)	90%
Solar intensity(I_s)	600 W/m^2
Sun angle of incidence(i)	10°
Lifetime degradation(D_L)	15%

The array capacity needed at the Beginning of Life (BOL) is equal to

$$BOL = \frac{EOL}{(1 - D_L)(\cos i)E_t} = 265\ W \quad [1]$$

The required surface area of solar cells is

$$A = \frac{BOL}{I_s * E} = 3.9\ m^2. \quad [2]$$

The approximate weight of the solar cells in kg is 0.04*BOL=10.6 kg [Sp 2;319]. We must add 14 kg for the solar panel deployment mechanism, and 20 kg (4% of orbiter dry weight) for wiring, for a total weight of 44.6 kg.

If an RTG were to be used, ten RTG Modules would be needed to provide 200 Watts of power. Ten RTG Modules would have a weight of 22 kg and a lifetime of eight years [Sp 3;sec.7,25]. However, the RTG would generate approximately 2700 W of waste heat [Sp 3;sec.7,25]. It would require over 5.5 square meters of "perfect" radiator surface (that is, a surface with an emissivity of 1 that did not absorb any heat from the environment) to dissipate this heat at an acceptable operating temperature [Sp 3;sec.10,8]. This would greatly increase the mass of the orbiter.

A decision matrix was created for the power system trade study. Each aspect of the power systems was given a weighting factor of 0 to 4, and the Solar cells were compared to the RTG and also rated from 0 to 4 in each aspect.

For a power output of 200 W, solar panels would weigh 25 kg while an RTG would weigh 22 kg. Therefore, solar panels were given a weight rating of 3 and RTGs were rated at 4.

Solar cells would cost approximately $50000. RTG modules would cost a total of approximately 4 million dollars and they are extremely difficult to acquire [Sp 3;sec.7,25]. For cost, solar was rated at 4 and RTG was rated at 1. For 200 W, solar cells have a size of 12.25 m^2 compared to approx. 2 m^3 for the RTG [Sp 3;sec.7,25]. For size, solar was rated at 2 and RTG was rated at 4. As previously described, the RTG has a problem with waste heat. For thermal control, solar was rated at 4 and RTG was rated at 2. All of the ratings and weights were input to the decision matrix. Each rating was multiplied by the corresponding weight and the products were added together for the solar panels and the RTGs, as shown in Table 3.3.5-4.

Table 3.3.5-4: Power System Decision Matrix

Aspect	**Solar**	**RTG**	**Weight Factor**	**Solar (Weighted)**	**RTG (Weighted)**
Mass	3	4	1	3	4
Cost	4	1	2	8	2
Size	2	4	3	6	12
Thermal Control	4	2	3	12	6
Total				**29**	**24**

From Table 3.3.5-4, it can be seen that solar panels should be used as the main power source for the orbiter.

The Common Spacecraft Bus uses silicon solar panels and NiH_2 batteries to provide power. The power output of the CSB solar panels[Sp 1;4] must be scaled for the changes in the mission. Table 3.3.5-5 shows CSB power production capabilities for Martian orbit.

Table 3.3.5-5: Common Spacecraft Bus Power Production

Power Requirement (CSB mission)	267.5 W
Distance Factor (1.5 AU)	$1/1.5^2$
Reduction in degradation (9% rather than 43%) [Sp 1;8]	1.60
Total Power Supplied	**190 W**

Since the orbiter has a power budget of 190 W, the CSB power system is sufficient for the needs of the orbiter.

The orbiter uses NiH_2 batteries for periods when it is not exposed to sunlight. These batteries must provide 51 watts of power for 8 hours of operation. Because of the estimated 10,000 battery cycles which will occur during the lifetime of the orbiter, the battery has a discharge depth of 50%, which doubles the battery mass [Sp 2;318]. The mass of the battery has been estimated to be 18 kg [Sp 2;319].

3.3.6 Orbiter Thermal Control

The spacecraft thermal control will be maintained by passive methods built into the Common Spacecraft Bus. These methods are multi-layer blankets, coating, optical layer reflectors, and heaters. The CSB thermal control system is able to keep an instrument cooled to 108 K and has been verified by PC-ITAS heat transfer software[Sp 1;9]. The surfaces mounting the solar panels have heat dissipation equipment and optical solar reflectors to radiate heat.

3.3.7 Orbiter Communications and Computation

The Mars-Silva spacecraft will all have nadir-pointing orientations for the duration of their mapping missions. With this positioning, the remote sensing instruments aboard each orbiter will be able to view Mars continually for the course of the project. The spacecraft will support the data collection of the instruments with onboard bubble memory for data that cannot be immediately relayed back to Earth. There will be daily playbacks of all recorded data , along with some real-time data transmissions for experiments with high data rates, such as the High Resolution Camera. The transmit power amplifiers are traveling -wave tubes.

Communications to and from the orbiters are accomplished by using a low gain transmission antenna and two low-gain receive antennas for near Earth and near Martian operations as well as in case of emergencies. A two-axis articulated high-gain antenna will be utilized during the late-cruise and mapping phases.

The telecommunications system will provide each orbiter with X-band communications compatible with the Deep Space Network. Communications will be both to and from Earth for tracking, telemetry, commanding, and data relay.

The orbiter will use a Rockwell RI-1750A/B computer. It uses a military-standard 1750 Instruction Set Architecture with a 16-bit processor, a throughput of 1.8 Million instructions per second, and 3.9 Megabytes of memory. It has a weight of 2.5 kg and a power requirement of 6.6 Watts.

3.4 Probe Deployment Module Design

The spacecraft design process is circular and required many iterations to arrive at the present configuration. Appendix Sp-B illustrates the flow of design parameters in the design of the spacecraft. Originally, the PDM mass was assumed to be 250 kg and the design process was followed through until it was discovered that the PDM mass would actually be much smaller. Finally, the spacecraft mass and size requirements were determined.

3.4.1 Probe Deployment Module Sizing

It will be shown in Section 3.4.3 that the spacecraft mass, with the propulsion system and propellant, is approximately 1100 kg. To establish the volumetric constraints it should be noted that the Proton fairing envelope has a diameter of 3.3 meters and a length of 7.5 meters[Sp 2;675]. According to the Probes Element, the canisters containing the micro-rovers have a width of 1 m and a height of .6 meters and the penetrators mounted on the Probe Deployment Module have a length of 1.3 meters.

Therefore, the PDM has a length of 1.3 meters and a width of 2 meters. The orbiter has a length of 1.2 meters, a width of 1.2 meters, and a mass of 250 kg. The engine has a mass of 550 kg distributed inside the PDM, the probes have a total mass of 250 kg and the PDM has a mass of 8 kg. This information is necessary for a simple moment of inertia calculation. Assuming point masses, the moment of inertia about the axis of rotation is:

$$I_z = \frac{1}{2} m_{engine} r_{PDM}^2 + m_{PDM} r_{PDM}^2 + m_{probe} r_{probe}^2 + m_{CSB} r_{CSB}^2 = 840 \text{ kg} \bullet \text{m}^2. \quad [1]$$

The moments of inertia about the other axes are

$$I_y = m_{engine \& PDM} \left(\frac{L_{PDM}^2}{3} + \frac{r_{PDM}^2}{4} \right) + \frac{m_{probe} L_{probe}^2}{3} + m_{CSB} \left(\frac{L_{CSB}^2}{3} + \frac{r_{CSB}^2}{4} \right) 737 \text{ kg} \bullet \text{m}^2 = I_x. \quad [2]$$

Without the orbiter, I_z=750 kg•m^2 and I_y=585 kg•m^2. Therefore, if either the Probe Deployment Module or the entire spacecraft begins to rotate, they will eventually rotate about the z axis.

3.4.2 Probe Deployment Module Structural Analysis

The design of the probe deployment module is driven by the launch conditions and the size of the probes to be deployed. The spacecraft will be launched into space on a Proton rocket although it can also be launched on an Atlas rocket. The maximum loading conditions during launch and flight, summarized in Table 3.4.2-1, are modeled as a load of 7 times the spacecraft weight (1100 kg at a gravity of 9.81 m/s^2) in the axial direction and 1.6 times its own weight in the lateral direction [Sp 2;687]. The maximum bending moment is found by multiplying the maximum lateral force by the maximum distance from the center of gravity.

Table 3.4.2-1: Vehicle Launch/Flight Load Conditions

Axial	76,000 N
Lateral	17,500 N
Bending moment (Lateral*half-length)	21,600 N•m
Equivalent axial= $P_{axial} + \frac{2M}{R}$	120,000 N

Due to the uncertain nature of the vehicle's true structural integrity, the equivalent axial load must be multiplied by a safety factor. Since multiple spacecraft will be constructed and only one will be tested, we apply a safety factor of 1.25 for yield conditions and 1.5 for ultimate conditions [Sp 2;439]. This results in an axial yield load condition of 150,000 N and an ultimate axial load condition of 180,000 N.

Because the CSB is made of 6061-T6 Aluminum [Sp 1;9], we will also use this material for simplicity. The material properties for 6061-T6 Aluminum are summarized in Table 3.4.2-2 [Sp 2;347]

Table 3.4.2-2: Material Properties of 6061-T6 Aluminum

Modulus of Elasticity	$68*10^9$ N/m^2
Poisson's Ratio	0.33
Density	2710 kg/m^3
Ultimate tensile strength	$290*10^6$ N/m^2
Ultimate yield strength	$240*10^6$ N/m^2

The Proton has a fundamental frequency of 30 Hz in the axial direction and 15 Hz in the lateral direction. The Atlas has lower fundamental frequencies, 15 Hz axial and 10 Hz lateral. This means that the first fundamental frequency of the spacecraft must be greater than that of the Proton launch vehicle.

Because the Common Spacecraft Bus was designed for a Pegasus launch vehicle, its lateral frequency condition of 18 Hz will be assumed [Sp 2;688]. The vibration conditions are summarized in Table 3.4.2-3.

Table 3.4.2-3: Launch Vehicle Fundamental Frequencies

Proton	30 Hz axial; 15 Hz lateral
Atlas	15 Hz axial; 10 Hz lateral
Pegasus	18 Hz axial; 18 Hz lateral
Worst-case conditions	30 Hz axial; 18 Hz lateral

The minimum PDM cross-sectional area to satisfy the axial frequency condition is

$$A = \frac{f_{nat}^2 mL}{0.0256E} = 0.00142 \text{ m}^2, \qquad [3]$$

requiring a thickness of 0.023 cm[Sp 2;454]. The minimum area moment of inertia is

$$I = \frac{f_{nat}^2 mL^3}{0.076176E} = 0.00108 \text{ m}^4 \qquad [4]$$

to satisfy the lateral vibration condition. This requires a thickness of

$$t = \frac{I}{\pi R^3} = 0.035 \text{ cm}. \qquad [5]$$

As the Proton ascends, the atmospheric pressure drops while the internal pressure remains constant and must be vented outside. A maximum pressure differential of 1 psi is assumed [Sp 2;686]. For a thickness of 0.035 cm, this creates a stress of

$$\sigma_p = \frac{pR}{t} = 20{,}000{,}000 \text{ N/m}^2. \qquad [6]$$

This reduces the effective yield strength of the material to 220*10^6 N/m^2 and the effective ultimate strength to 270*10^6 N/m^2.

For the axial yield load requirement of 150,000 N and a yield strength of $220*10^6$ N/m^2, the required area is 0.00068 m^2. For the ultimate load requirement of 180,000 N and an ultimate strength of $270*10^6$ N/m^2, the required area is 0.00066 m^2.

The thickness of the PDM skin is dependent on the structural condition that requires the maximum thickness which in this case is the lateral vibration of the launch vehicle. The lateral vibration condition requires a thickness of 0.035 cm for a PDM structural mass of 8 kg.

The critical buckling load for the cylinder is

$$P_c = \frac{\pi^2 EI}{L^2} = 118{,}000{,}000 \text{ N,} \qquad [7]$$

which is much more than the ultimate load condition of 180,000 N.

3.4.3 Spacecraft Mass

Adding together all the components which are summarized in Table 3.4.3-1, the spacecraft has a total mass of approximately 1100 kg. Each of the three spacecraft carries a slightly different payload and therefore has a different overall mass. A more detailed analysis of the probe mass budget follows in section 3.5.2.

Table 3.4.3-1: Total Spacecraft Mass

Orbiter	250 kg
PDM Structure	8 kg
PDM thrusters & propellant	35 kg
Probe deployment mechanisms	7 kg
Spacecraft propulsion system	550 kg
Probes (estimate)	250 kg
Total Spacecraft Mass Estimate	**1100 kg**

3.4.4 Probe Deployment Module Guidance, Navigation, and Control

The PDM also has a system of thrusters identical to the thruster system on the CSB. The system uses six hydrazine thrusters and propellant tanks with 20 kg capacities. Each thruster produces 0.89 Newtons of force. The thrusters are on the outside rim of the PDM and produce 0.89 N•m of yaw moment [Sp 1;8]. Argos Space Endeavors has determined that this thruster arrangement will provide the necessary additional GNC.

3.5 Spacecraft and Launch Configuration Design

The PDR 1 level Aeneas Project design called for only one spacecraft (Mars-Silva) and two probe deployment modules (Romulus and Remus). At the time, the Proton launch vehicle was the booster of choice due to its relatively low cost and high injection-mass capability. This concept is viable for mission success; however, Argos Space Endeavours has replaced the single-spacecraft / single-launch design with a multiple-spacecraft / multiple-launch design. The primary reason for this change in mission concept was that the initial design had two major single-point-failures. The mission would have had 0% science return if either the booster or spacecraft failed. In light of the recent Mars Observer failure, political and scientific pressures have caused NASA to turn to missions which will provide some level of mission success in the event of system failure. Therefore, Argos Space Endeavours decided to provide redundancy at both points with multiple spacecraft and multiple launches. The trade-off for higher mission success rate, cost, and increased complexity, are investigated in the following subsections.

3.5.1 Multiple Spacecraft Configuration

Once the multiple spacecraft concept was initiated, a new question emerged: How many spacecraft are required to provide adequate redundancy? Since Mars is the primary objective and Phobos is a secondary objective of the mission, two spacecraft will be dedicated to deploying probes to Mars and a third spacecraft will be dedicated to deploying probes to Phobos. No more than three total spacecraft are chosen because of the self-imposed cost constraint of $500 million.

Each spacecraft consists of three main components: an orbiter, a probe deployment module (PDM), and a probe package. For the first iteration, the total mass allocation for each spacecraft was 750 kg (250 kg for each main component). This number was later revised to 550 kg once the PDM mass was determined to be only 50 kg. The next step was to select and size the probe package. The constraints for the probe package design were mass, redundancy for the individual probes, cost, and total volume of the spacecraft. The selection process of the different types of probes used can be found in section 4.6. The result of the selection process is listed in Table 4.7-1 in Probe Selection, Section 4.7. Other configurations considered are listed in Appendix P-D.

Once the total mass of each spacecraft was determined, the next step was to determine which engine to perform the 2^{nd} ΔV for the Mars orbit insertion maneuver (MOI). ΔV_2 was determined to be 2.1 km/s from trajectory analysis. Using the rocket equation and the maximum mass constraint of Mars-Silva 3, two engines were taken into consideration: the STAR 37XFP (solid motor) and the R-40B (bipropellant engine). Their characteristics are listed in Table 3.5.1-1.

Table 3.5.1-1: Engine Characteristics

STAR 37XFP Characteristics [Sp 5;10-15]	**R-40B Engine Characteristics** [Sp 6;646]
Manufacturer : Thiokol	Manufacturer : Marquardt
Nominal Length : 1.5024 m	Propellants : N_2O_4/MMH
Nominal Diameter : 0.9322 m	Engine Mass : 7.26 kg
Ignition Mass : 957.67 kg	Nominal Thrust : 4000 N
Burn-Out Mass : 63.64 kg	Isp : 309 sec
Average Thrust : 37,700 N	Operation Life : 25,000 sec
Effective Isp : 290 sec	
Burn Time : 66 sec	

The design of the STAR 37XFP is rigid and flown as is. The R-40B engine, on the other hand, is more flexible since the two propellant tanks can be sized and shaped as necessary to meet the volume constraint. Assuming that the propellant tanks are spherical, the sizing results are as shown in Table 3.5.1-2.

Table 3.5.1-2: Tank Sizing Results

Maximum propellant mass required	534.96 kg (from Mars-Silva 3)
Fuel to oxidizer mass ratio	2.17 (for Isp of 310 sec)
Density of N_2O_4	1447 kg/m^3
Density of MMH	878.8 kg/m^3
Mass of N_2O_4 required	366.203 kg
Mass of MMH required	168.7571 kg
Diameter of N_2O_4 tank	0.78481 m
Diameter of MMH Tank	0.7158 m

Furthermore, assuming that the propellant tanks are placed next to each other, the total diameter of the two tanks together is less than 1.5 m. With the probe deployment module having a dimension of 2.0X1.3 m, both tanks can be fitted within the PDM. Although the STAR motor may be fitted within the PDM also, it requires almost twice as much mass compared to the R-40B. In addition, the R-40B engine has the capability of multiple ignitions whereas the solid STAR motor does not have the restarting capability once it is fired. For these reasons, Argos Space Endeavours selected the R-40B engine (at a cost of about $10 million per engine) for MOI. Table 3.5.1-3 sums up the final spacecraft mass configuration. This configuration allows Mars-Silva 1 and 2 to be launched together on a single Proton and Mars-Silva 3 on a second Proton.

Table 3.5.1-3: Two Launch Spacecraft Mass Configuration

	LAUNCH NO. 1		LAUNCH NO. 2
Orbiter	MARS-SILVA 1	MARS-SILVA 2	MARS-SILVA 3
Mass (kg)	250	250	250
PDM	50	50	50
Probes	238.5	150	253.1
R-40B Engine (Dry)	7.26	7.26	7.26
Propellant	545.35	456.92	559.94
MOI Mass Required	1091.11	914.18	1120.30
MOI Mass (Actual)	1118.38	937.03	1148.30
	MS1 & MS2		**MS3**
Injected Total	**2055.41**		**1148.30**

Table 3.5.1-4: Launch Constraints

ΔV2 (m/s)	2100	
Isp2 (s)	309	**R-40B**
ΔV1 (m/s)	3600	
Isp1 (s)	350	**D-1e**
C3L (km^2/s^2)	8.8	
C3L Mass(kg)	**5400** [Sp 5;A21]	**Proton/D-1e**

3.5.2 Launch Configuration

The three spacecraft could be launched together on one launch vehicle or separately on two or even three launch vehicles. The constraints on the booster in any case are cost, payload volume, and injection-mass capability. The first option is to launch all three spacecraft on one booster. This would require a Titan IV and the Centaur upperstage; however, the cost of the Titan

IV/Centaur is over $400 million [Sp 5,8-7]. The next option is to launch the three spacecraft on three separate launch vehicles. This option requires three Atlas II boosters with centaur upperstage. This booster/upperstage combination costs approximately $80 million per launch giving a total of over $240 million [Sp 5,7-3]. Although both options are valid, they are quite expensive. Therefore, a third approach is taken. A single Proton Model C with D-1e upperstage has enough payload volume and power to inject Mars-Silva 1 and 2 together into the interplanetary trajectory. A Proton Model A with the D-1e upperstage can deliver the remaining Mars-Silva 3. The only difference between the two Proton models is the payload volume (see Appendix Sp-A). Each Proton/D-1e combination costs about $40 million [Sp 5,A-9] giving a total mission launch cost of $80 million. Thus due to the relatively high launch costs associated with the Titan IV and Atlas II scenarios, Argos Space Endeavours has selected the Proton launch configuration for Project Aeneas. The first launch configuration (Mars-Silva 1 & 2) is illustrated in Figure 3.5.2-1.

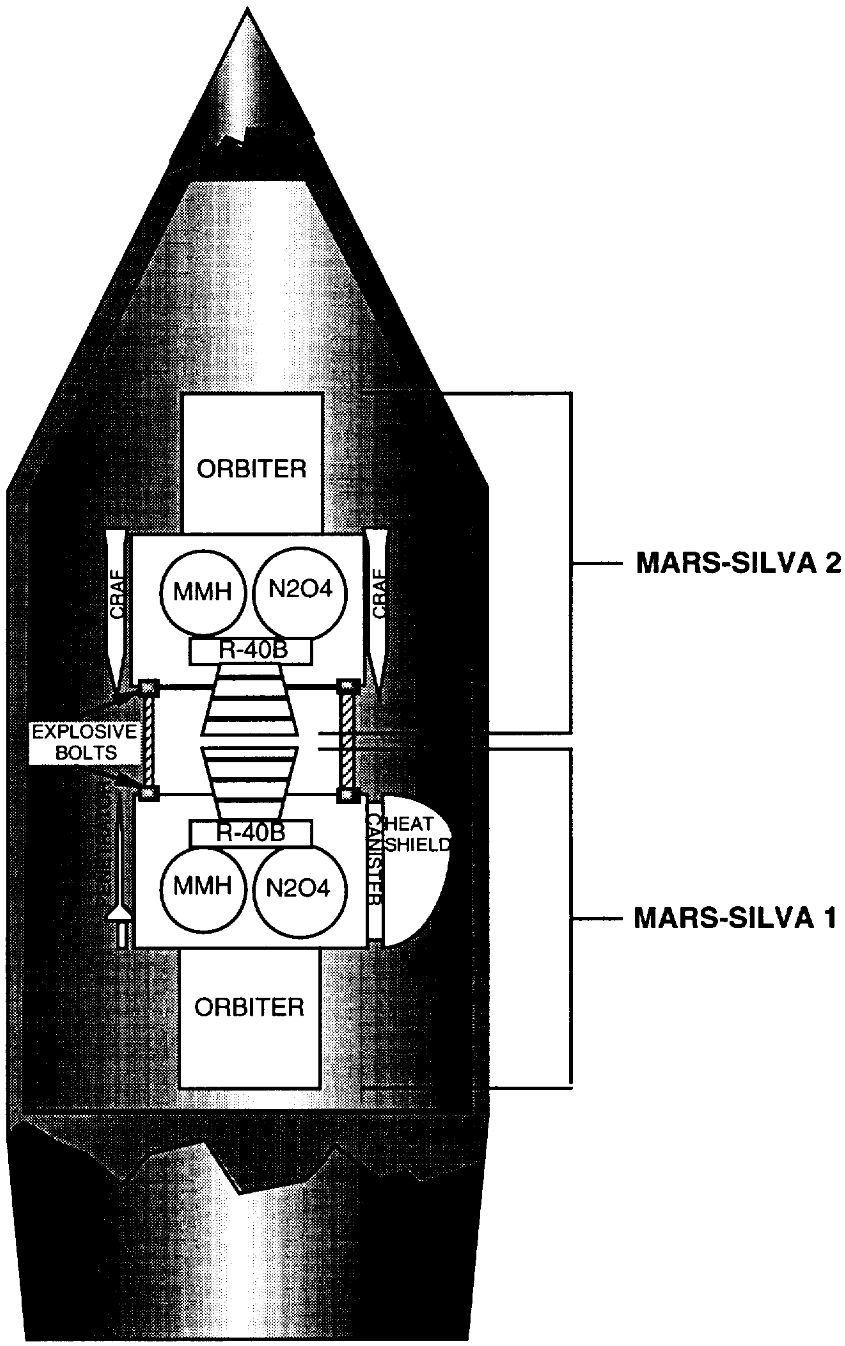

Figure 3.5.2-1: Proton Launch Configuration for Mars-Silva 1 & 2

3.5.3 Spacecraft & Launch Vehicle Cost Estimates

From the beginning of Project Aeneas, Argos Space Endeavours had a self-imposed project cost constraint of $500 million. This project cost constraint later became one of the driving factors in the decision making process for probe packages, spacecraft, and launch vehicles. The cost outcome for the project is summarized below.

Table 3.5.3-1: Total Mission Cost Summary

3 Mars-Silva spacecraft	$380 million
3 R-40B engines @ $10 million per engine	$30 million
2 Proton/D-1e @ $40 million each	$80 million
SUBTOTAL	$490 million
+ 2% margin	$9.8 million
TOTAL	**$498.8 million**

4.0 Probe Element

The key requirement for successful Mars exploration is the establishment of long term science stations at diverse Martian locations in order to conduct seismic, meteorological, and geoscience experiments. These robotic explorations will provide information necessary to increase the success and safety of future piloted missions. The Probe element has considered various types of probes which are capable of fulfilling the above requirement for Project Aeneas. These probes include balloons, penetrators, landers, canisters, rovers, and micro-rovers. Also, an ISRU probe, which will be used as a proof of concept, has been developed. Trade studies have been performed and analyzed to determine the optimal probe combination for Project Aeneas. It has been determined that the best combination of probes consists of penetrators and canisters with micro-rovers. Also, we will use Comet Rendezvous Asteroid Flyby (CRAF) technology for the investigation of Phobos. The selected probes will assist in fulfilling the goals of Project Aeneas.

The goals of the Probe Element include technology, science, and mission experiments. A technology experiment is an experiment where equipment is tested (in our case the equipment would be the probes) to see if it functions as intended. One way to demonstrate a technology experiment is to perform a science experiment. A science experiment is an experiment where information about a given object or condition is gathered. For example, determining the elemental composition of the Martian regolith using the instruments on the probes would be a science experiment. Mission experiments incorporate both technology and science experiments and answer the question "How is our technology going to be used to perform the science experiments?". The mission experiment Argos Space Endeavors is performing is Project Aeneas.

4.1 Penetrator

Penetrator type probes are those which pierce a planetary surface embedding themselves in the local regolith during the process of impact.[P 1;10] It is a low complexity, low cost option to sample numerous, widely separated planetary locations.[P 1;3] Although the penetrator is a stationary probe, its relatively low mass and small dimensions (20 kg, length =1.35 m) allows the use of several penetrators.[P 1;94, 13][PA; Fig. PA-1] Multiple penetrators improve mission success through redundancy. Also, the collection of simultaneous data from distant sites yields more scientific insight than a single data source.[P 1;90]

The penetrator is deployed from the orbiter and releases an aerobraking system upon atmospheric entry for deceleration. This deceleration protects the penetrator from excessive heating and aerodynamic loads.[P 1;4] The use of passive decelerators (aeroshell and parachute) eliminates the need for a propulsion system. Also, there is no need for a guidance and control system since the penetrator's aerodynamic surfaces are sufficient to guide it to the surface, even in Mars' thin atmosphere.[P 1;10] Penetrators utilize a support system located on the aft section to prevent it from becoming completely buried (see Fig. 4.1-1) The aft section begins with a conical flare that doubles the penetrator diameter.[P 1;40] Once the support makes contact with the surface; the penetrator separates into two sections and the fore section will penetrate deeper into the Martian surface (see Fig. 4.1-1). The total anticipated penetration in hard and soft soil is 1.65 m and 4.08 m respectively (Refer to [P 1;42] for a more detailed analysis of penetrator emplacement).[P 1;49, 50]

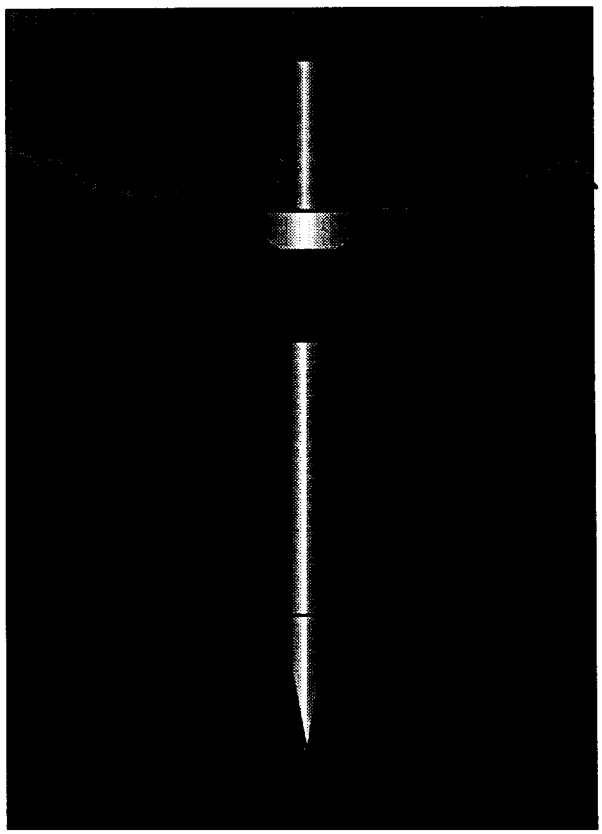

Fig 4.1-1: Embedded Penetrator

Current penetrator designs can withstand loads of up to 500 g's at surface contact and velocities of about 80 m/s.[P1; 54][P-A Table PA-1] Aluminum honeycomb regions that crush during impact absorb some of the impact energy thereby protecting the instruments.[P1; 52] These instruments include accelerometers, low-power spectrometers, seismometers, and small meteorological packages. The data from the penetrator is transmitted with a low-gain antenna to an orbiter or a surface station. These can then relay the data to Earth.

4.2 Lander

Small landers are intended for surface meteorological measurements as well as magnetic field and seismic measurements. The lander chosen for Project Aeneas was a canister type lander capable of carrying 2 micro-rovers. The lander will communicate to an orbiter which will relay data back to Earth. It can also communicate directly to Earth with a low-gain antenna for redundancy; however, this lowers the data rate. The lander will serve as a communication relay for the micro-rover since the rover does not have the power to communicate to the orbiter.

Landing on the Martian surface entails atmospheric entry behind a heat shield, deploying a parachute, and crashing into the ground atop energy absorbing structures or airbags that inflate just before impact. Current landers can withstand an impact of 200 g's at a surface contact velocity of about 10 m/s.[P5] Once on the ground, the lander will deploy two ramps to allow each micro-rover to exit (see Fig. 4.2-1)

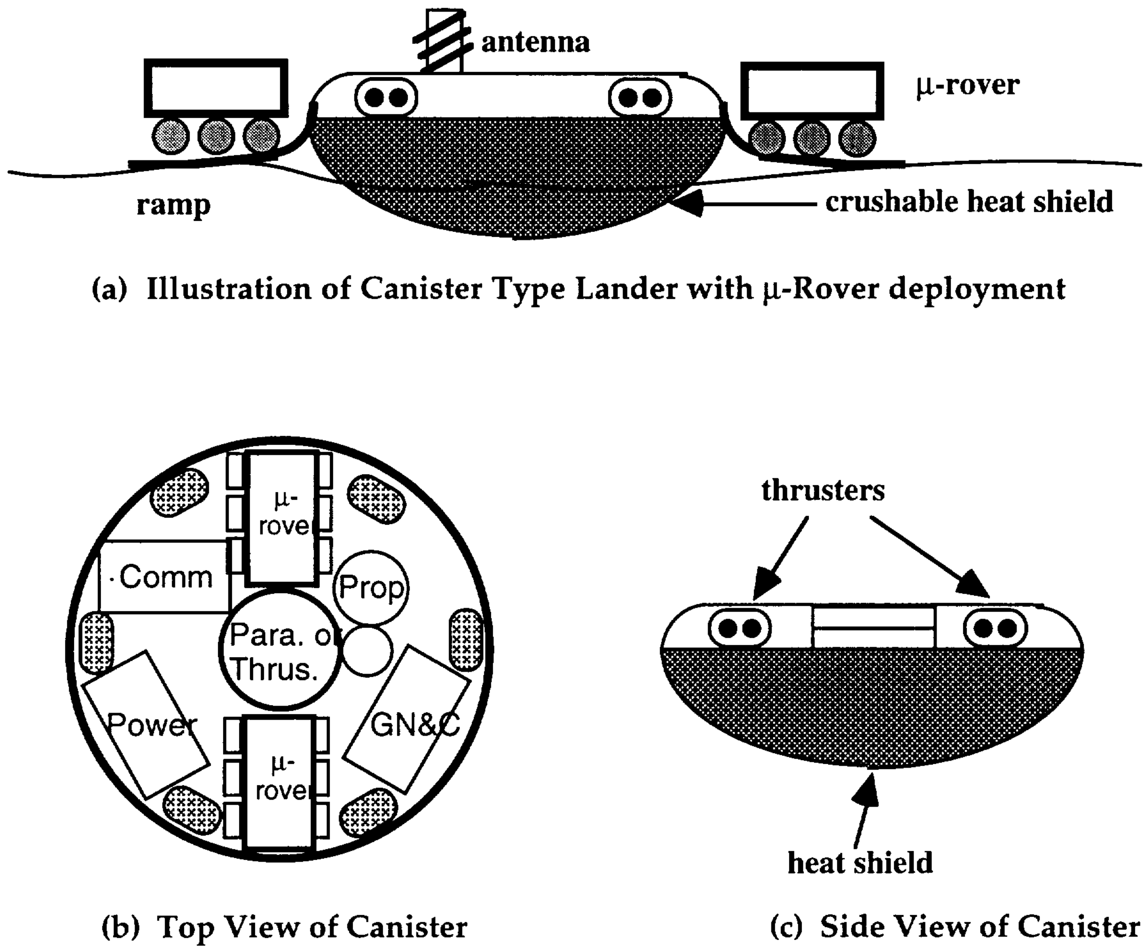

Figure 4.2-1 (a-c): Prototype Lander Concept

4.3 Micro-rover

The micro-rover chosen for Project Aeneas is JPL's Rocky IV(see Fig. 4.3-1). Rocky IV is roughly the size of a desktop computer (length = 60 cm, width = 46 cm, height = 28 cm.) with a mass of 8 kg.[P 2; 4][PB; Fig. PB-1] There will be two micro-rovers per canister. Rocky IV uses a Motorola RF Modem with a range of 1.9 km at 9600 baud.[P 2; 4] It is not capable of communicating directly to Earth or to an orbiter and hence needs a small lander to relay data.

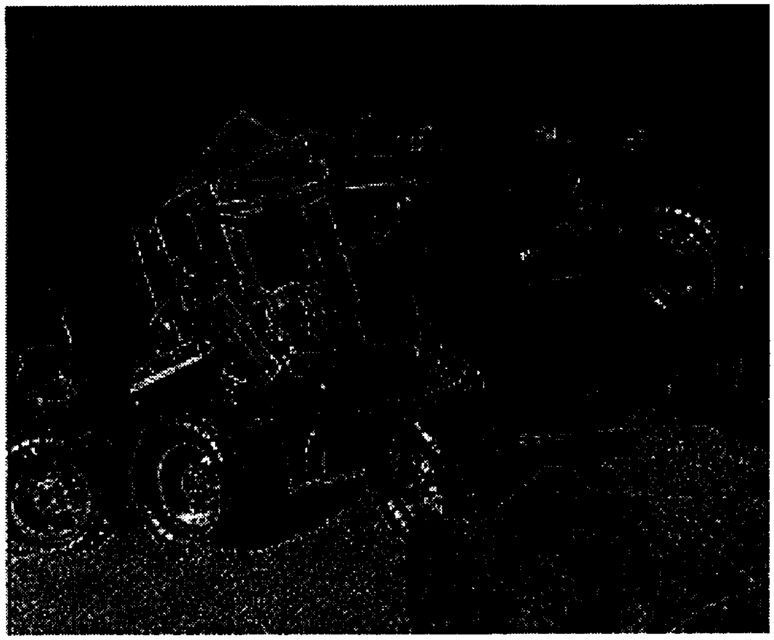

Figure 4.3-1: JPL's Rocky IV μ-Rover

Rocky IV is a remote controlled rover capable of carrying single scientific packages such as an Alpha Proton X-ray Spectrometer (APXS). Also, the micro-rover's mobility provides exploration at different locations around the landing site. Once Rocky IV is on the Martian surface it will image the terrain with stereo vision cameras and send the picture back to Earth. Scientists wearing 3-D goggles will plot a course for the micro-rover based on the image .[P 3; 15] Rocky IV has proximity detectors which can sense large obstacles that will help negotiate a new route using a series of IF-THEN statements. If this fails, the rover will signal for a new route. Rocky IV is capable of traversing a maximum of 20 m/day. However, this is not likely since it has to remain stationary to perform scientific experiments which can take up to 10 hours.[P 3; 11]

Rocky IV has six wheels each powered by a 2-Watt motor. The rover's tires are steel because rubber would crumble at Martian temperatures.[P 2; 5] It is either battery or solar powered. The battery pack has an output of 150 W hr and the solar-power array, which is 2000 sq. cm, has an output of 100 W hr/day on a clear day and 50 W hr/day in a dust storm.[P 2; 4][PB; Table PB-1]

4.4 ISRU probe

The ISRU fuel production experiment probe will serve as a technology proof of concept. It will gather Martian carbon dioxide from the atmosphere which will be used to produce methane which can be used as a fuel. Refer to the Science Element report for a detailed description of the requirements and key objectives of the ISRU probe.

4.5 Phobos Probe

The Phobos probe will be based on the work which has been carried out for NASA's Comet Rendezvous and Asteroid Flyby (CRAF) mission and the European Space Agency's (ESA) Rossetta mission. Both of these missions were designed to deliver probes to bodies with negligible gravity fields (e.g. a comet, asteroid, or small moon). The key technology elements from the CRAF and Rossetta probes will be utilized to minimize the development time and cost of the Phobos probe.

1. Gamma ray spectrometer
2. Differential scanning calorimeter
3. Evolved gas analyzer
4. Aft thermal probe
5. Battery
6. Electronics module
7. Accelerometers
8. Thrusters
9. N2H4 tank
10. Mid thermal probe
11. Sample scoop
12. Fore thermal probe
13. Accelerometers

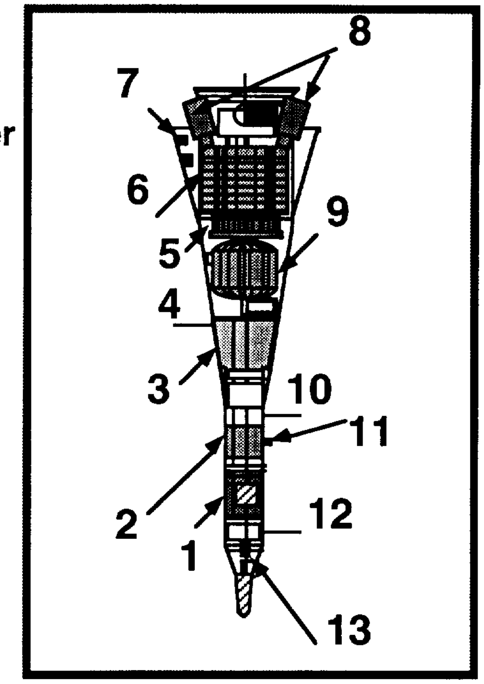

Figure 4.5-1: Breakdown of CRAF-type Phobos Probe

The Phobos probe is a penetrator-type probe (see Fig. 4.5-1). It will take surface measurements to determine the mineralogical, molecular, and elemental composition of Phobos as well as its thermal diffusivity and strength [P 4; 2]. The Phobos probe has a mass, power requirement, and maximum output data rate of 66.8 kg, 12.8 W, and 2000 bps, respectively [P 4; 21][PC; Table PC-1]. It will communicate to an orbiter which will relay data to Earth. The Phobos probe is capable of carrying a variety of instruments such as a gamma ray spectrometer, differential scanning calorimeter, and evolved gas analyzer (Refer to reference [P 4] for a detailed description of the instruments) [P 4; 16].

4.6 Probe Technology Trade Study

4.6.1 Preliminary Trade Studies

The probe element has performed a series of trade studies to determine the optimum combination of probes for the mission. The probe element distributed technical surveys to the members of Argos Space Endeavors and has used this feedback to determine "weights" for each probe attribute. A scale of 0 to 4 was used with 0 being the worst and 4 being the best. The Probe Element and the Science Element were given an initial weighting factor of 1.25 : 1 since it was determined that these two elements should have the most influence on the design and selection of the probe mission.

Table 4.6.1-1: Probe Technology Survey

	Element:	Probes	Science	Spacecraft	Trajectory	Contributing weighting factor
	Weight:	1.25	1.25	1	1	
Element Attributes						*TOTAL*
Mass		4	3	4	4	4.1875
Coverage		3	2	2	3	2.8125
Range		3	3	2	3	3.125
Power Requirement		3	3	2.5	2	3
Redundancy		3	3	2.5	1	2.75
Lifetime		3	3	2.5	2	3
Science Gathering		3.5	4	3	4	4.09375
Instrumentation		3	4	3	4	3.9375
Communications		3.5	4	4	4	4.34375
Cost		3	2	3.5	4	3.4375

Table 4.6.1-2: Probe Technology Decision Matrix

Probe Type:	μ-Rover	Rover	Penetrator	Balloon	Lander
Probe Attributes					
Mass	4	1	3	0	3
Coverage	1	4	0	3	0
Range	2	4	0	3	0
Power Requirement	4	0	4	2	3
Redundancy	2	2	2	2	2
Lifetime	2	3	2	2	4
Science Gathering	2	4	2	3	2
Instrumentation	2	4	3	3	3
Communications	2	4	3	3	4
Cost	4	0	4	2	3
TOTAL					
Unweighted	25	26	23	23	24
Weighted	87.8125	91.9375	82.84375	79.3125	86.75

The contributing weighting factors determined from the Probe Technology Survey were applied to the probe attributes in the probe technology decision matrix . The decision matrix yielded values which placed the rover as the "best" probe, followed by the micro-rover, lander, penetrator, and balloon. In addition, a set of normalized trade studies were performed with the addition of a canister which is a lander type probe capable of carrying two micro-rovers.

4.6.2 Normalized Trade Study

Since the results from the above trade study were not as conclusive as desired, two other trade studies were performed. The first trade study determines how many probe types could be obtained given a mass allocation 500 kg. All values are normalized with respect to the micro-rover values. This means that for each characteristic, the micro-rover was assigned a value of 1 and all other probes were assigned corresponding values. For example, if the micro-rover's lifetime is 1 month and the lander's lifetime is 12 months, then the corresponding values are 1 and 12 respectively. For each probe type, the instrument and range values were added together. The sum was then multiplied by the quantity and lifetime of the probe. The rational behind this method was that instrument and range together contribute only a factor to the overall science while the number of probes and their lifetime each contribute a factor to the overall science obtained. The results, called "factor", show that the canister, rover, and penetrator were the better choices. (Table 4.6.2-1)

Table 4.6.2-1: Science per 500 kg

	μ-Rover	Lander	Canister	Rover	Penetrator	Balloon
Quantity	50	6.4	7.14	3.6	18	6.3
Instrument	1	3.1	4.1	7	3.9	10
Range	1	0	2	25	0	7
Lifetime	1	12	12	7	12	0.25
Factor	100	238	523	806	842	27

Similarly, another trade study was performed with cost as the driving factor. Table 4.6.2-2 shows how much science can be obtained with $100 million allocated for each probe type. Here, the mass factor also comes into play as the "inverse factor". This means that the "mass factor" will be divided instead of multiplied (the minus sign here indicates the inverse factor and not a negative value). The results indicate that the penetrator, micro-rover, and canister are the best choices.

Table 4.6.2-2: Science per $100 Million

	μ-Rover	Lander	Canister	Rover	Penetrator	Balloon
Quantity	40	4.4	3.3	1.3	13.3	3.3
Instrument	1	3.1	4.1	7	3.9	10
Range	1	0	2	25	0	7
Life	1	12	12	7	12	0.25
Mass Factor	-1	-7.2	-7	-12.5	-2.77	-5.9
Factor	80	23	35	24	225	24

Based on the trade studies conducted by Argos Space Endeavors, it was determined that the best combination of probes consists of penetrators, canisters, and micro-rovers. Although penetrators are stationary probes, the use of multiple penetrators provides a wide coverage area. Multiple penetrators can be used because of their relatively low mass and volume. The micro-rover cannot be deployed independently since it has limited communication. Therefore, each canister will contain 2 micro-rovers, serving as a delivery system and communication relay for the micro-rover.

Although the rover faired the best in the decision matrix it was not chosen because of mass, power, volume, and cost constraints. The rover uses a Radio-isotope Thermoelectric Generator (RTG) as a power source which was not acceptable for Project Aeneas because of planetary quarantine regulations. The balloon was not chosen because of mass and volume constraints. Another reason why the balloon was not chosen was because it would not yield controlled results, like the other types of probes, since it is at the mercy of the Martian wind.

4.7 Probe Packages

Once the normalized trade studies have reduced the choices of probes, the next step is to design various probe packages or combinations. These packages must satisfy the following constraints for each spacecraft:

1. Total mass per probe package must be less than 250 kg.
2. Total cost of each package must be under $150 million (including orbiter).
3. Each probe type must have more than 1 probe to provide redundancy.

Various probe packages are proposed and listed in Appendix P-D. The chosen package is listed in Table 4.7-1.

Table 4.7-1: Probe Combinations Chosen for Project Aeneas

	Quantity	Mass (kg)	Cost (mil$)
		Mars-Silva 1	
Penetrator	5	138.5	37.5
ISRU	1	30	10
Canister	1	70	30
Orbiter	1	250	50
SUBTOTAL		**488.5**	**127.5**
		Mars-Silva 2	
CRAF	2	150	60
Orbiter	1	250	50
SUBTOTAL		**400**	**110**
		Mars-Silva 3	
Penetrator	3	83.1	22.5
ISRU	1	30	10
Canister	2	140	60
Orbiter	1	250	50
SUBTOTAL		**503.1**	**142.5**
TOTAL		**1391.6**	**380**

5.0 Orbits and Trajectories Element

The Orbits and Trajectory Element focused on the fulfillment of two primary goals. The first was to develop a launch, transfer orbit, and Mars orbit scenario which would satisfy the requirements of Project Aeneas. The second, to develop these orbits in such a way that they would satisfy the constraints imposed on them by orbital mechanics considerations.

The initial phase of the orbit design was carried out using Hohmann transfer approximations. This type of interplanetary transfer is a good approximation since the Earth-Mars geometry lends itself well to such analyses. The Hohmann transfer uses the assumption of zero plane change and 180° transfer (from periapsis at Earth to apoapsis at Mars). The orbit of Mars is inclined at 1.85° [OT 1] with respect to the Ecliptic plane. As a result, the small plane change ΔV needed to carry out an Earth to Mars transfer is minimal.

The Orbits and Trajectory Element then carried out the orbit design process beyond that of simple Hohmann transfers. An analysis of the launch energy as a function of departure and arrival dates allowed the group to determine the minimum launch energy transfer for the year 2002/3 launch opportunity. This analysis was carried out by solving Lambert's problem for the launch and arrival dates at Earth and Mars respectively. The resulting trajectories are very similar to the Hohmann transfer but reflect the actual trajectory constraints imposed by the orbital geometry of Earth and Mars. A total of 4 trajectories have been developed for the mission. All of the trajectories have a launch and arrival dates in the year 2003. Two of these trajectories were designed using the aforementioned criterion (minimum launch energy) while the other two were designed by minimizing the arrival energy. This was done in order to allow flexibility in the choice of launch vehicle and spacecraft mass allocation.

The final step in the trajectory design procedure is to choose the booster/upper-stage combination which will satisfy the launch criterion. A variety of booster/upper-stage combinations were evaluated in terms of injection mass as a function of launch energy. The Hohmann analysis, and the subsequent Lambert targeted trajectories, defined the launch energy requirements for an interplanetary transfer from Earth to Mars. Given this information it was possible to determine which combinations satisfied the launch energy requirement and the spacecraft mass requirement.

5.1 Interplanetary Trajectory

5.1.1 Hohmann Transfer and Patched Conic

The Orbits and Trajectory Element carried out preliminary design of the interplanetary trajectory using a simple Hohmann transfer [OT 4;61]. The simple Hohmann transfer assumes that there is no plane change. In addition, the Hohmann transfer assumes that the trajectory occurs from periapsis to apoapsis, resulting in a true anomaly change equal to 180°. Figure 5.1.1-1 illustrates the Hohmann interplanetary transfer trajectory.

A patched conic approach is used to compute the transfer ΔV's. Assuming that the Earth and Mars are in circular orbits around the Sun we compute the required periapsis and apoapsis velocities (relative to the Sun) for the spacecraft. The spacecraft is in a conic transfer orbit with respect to the Sun and in a hyperbolic orbit with respect to Earth and Mars. The task is then to compute Earth relative and Mars relative trajectories which will give the spacecraft the required hyperbolic excess velocities needed to match the apoapsis and periapsis velocities of the Hohmann transfer. The patched conic is thus composed of 3 trajectories: a hyperbolic orbit with respect to Earth which gives the spacecraft the required transfer periapsis velocity, the Hohmann interplanetary transfer, and a hyperbolic orbit with respect to Mars which matches the arrival apoapsis velocity.

The first ΔV occurs at Low Earth Orbit (LEO). This ΔV places the spacecraft in a hyperbolic trajectory relative to the Earth with a given hyperbolic excess velocity (V_∞). The value of V_∞ at Earth is equal the difference in the transfer periapse velocity and the Earth orbital velocity. A knowledge of the required V_∞ allows us to compute the ΔV at LEO. The second ΔV is performed at Low Mars Orbit (LMO) and is computed in the same way as the first ΔV. The launch and arrival energies are then computed as the square of the V_∞'s. This value, which is actually equal to twice the orbital energy, is denoted by the term C3.

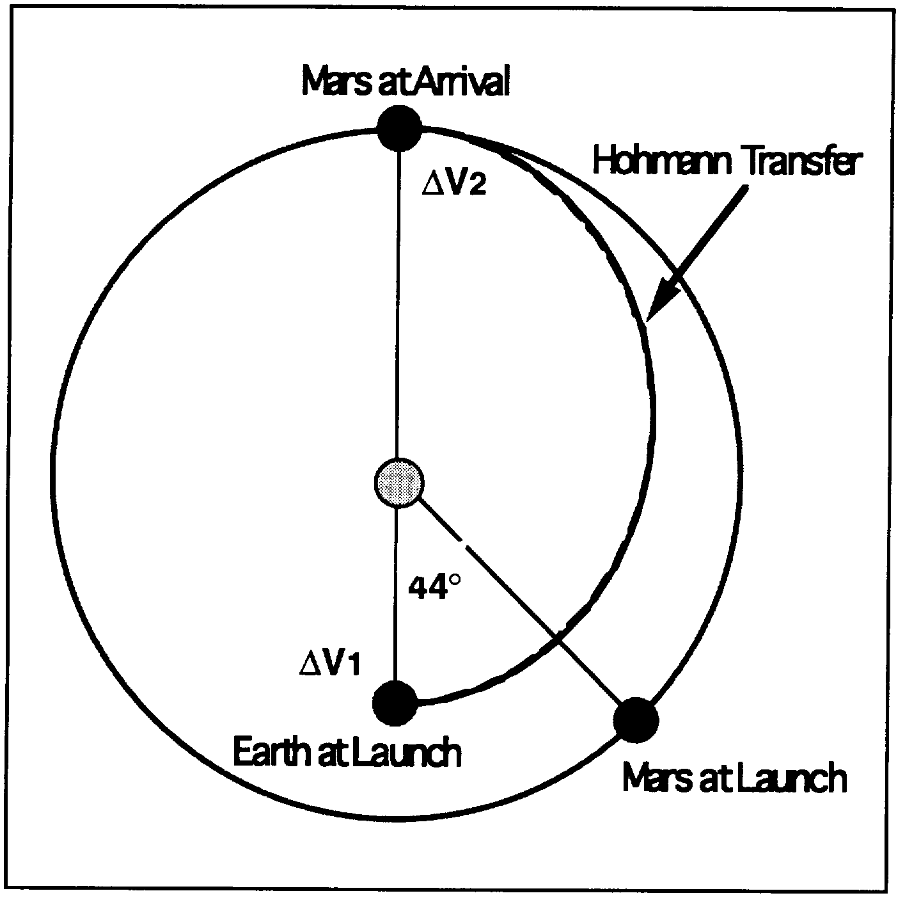

Figure 5.1.1-1: Hohmann Transfer to Mars

The Hohmann transfer calculations were performed using a TK Solver model [Appendix OT-A]. The parameters computed for the trajectory are presented below:

LEO Radius =	6578.145 km
LEO Altitude =	200 km
LMO Radius =	3727.20 km
LMO Altitude =	330 km
Phase Angle =	44.3°

$\Delta V1$ =	3.61 km/s
$\Delta V2$ =	2.09 km/s
Total ΔV =	5.70 km/s

V_∞ Launch = 2.94 km/s
V_∞ Arrival = 2.65 km/s
C3 Launch = 8.67 km2/s2
C3Arrival = 7.02 km2/s2

5.1.2 Lambert Targeting

The value of launch C3 will be the driving factor in selecting an optimized orbit for the transfer. A purely Hohmann transfer is generally not possible to obtain because it would require Mars to be on the ecliptic plane at arrival. This requirement can only be satisfied when Mars is at one of its nodes (with respect to the Ecliptic plane). Furthermore, the phasing requirement of 44°, in conjunction with the node requirement, is a geometry which can only be satisfied once every 15 years. This period is clearly too long a wait for launching a mission if the window is lost.

As a solution to this problem we introduce Lambert targeted trajectories. Given the position vectors of the target bodies and the time of flight, it is possible to solve Lambert's problem and generate a trajectory which will meet these constraints [OT 4;92]. Thus, the Hohmann requirements of no plane change and 180° transfer no longer affect the design process.

Even though there is no particular restriction on the design of a Lambert trajectory there are still launch geometry issues which make certain launch dates more attractive than others in terms of launch energy requirements. The launch window for the Earth to Mars transfer based on launch energy requirements that can be met by current launch systems occurs approximately every two years (e.g. 1994, 1996, ...). The Orbits and Trajectory Element generated a contour plot of C3 versus launch and arrival dates for the 2002/3 launch opportunity (Appendix OT-B). This was done by solving the Lambert problem for the Earth to Mars transfer over the range of launch and arrival dates. This plot, also known as a "pork-chop" plot, was then used to identify a small region of launch and arrival dates with minimum launch C3's. This range of global minima was then inspected using QUICK [OT 1] to identify the overall minimum launch C3 (and it's associated launch and arrival dates) for the 2002/3 launch opportunity. Figure 5.1.2-1 contains a plot of the Lambert targeted Earth-Mars transfer trajectory. The parameters of this trajectory are:

Type:	I
Semimajor Axis:	188444700 km
Eccentricity:	0.19
Inclination:	0.14 deg
Launch Date:	7 June 2003
Arrival Date:	25 December 2003
Time of Flight:	201.7 days
Launch C3:	8.8095 km2/s2
Arrival C3:	7.3163 km2/s2

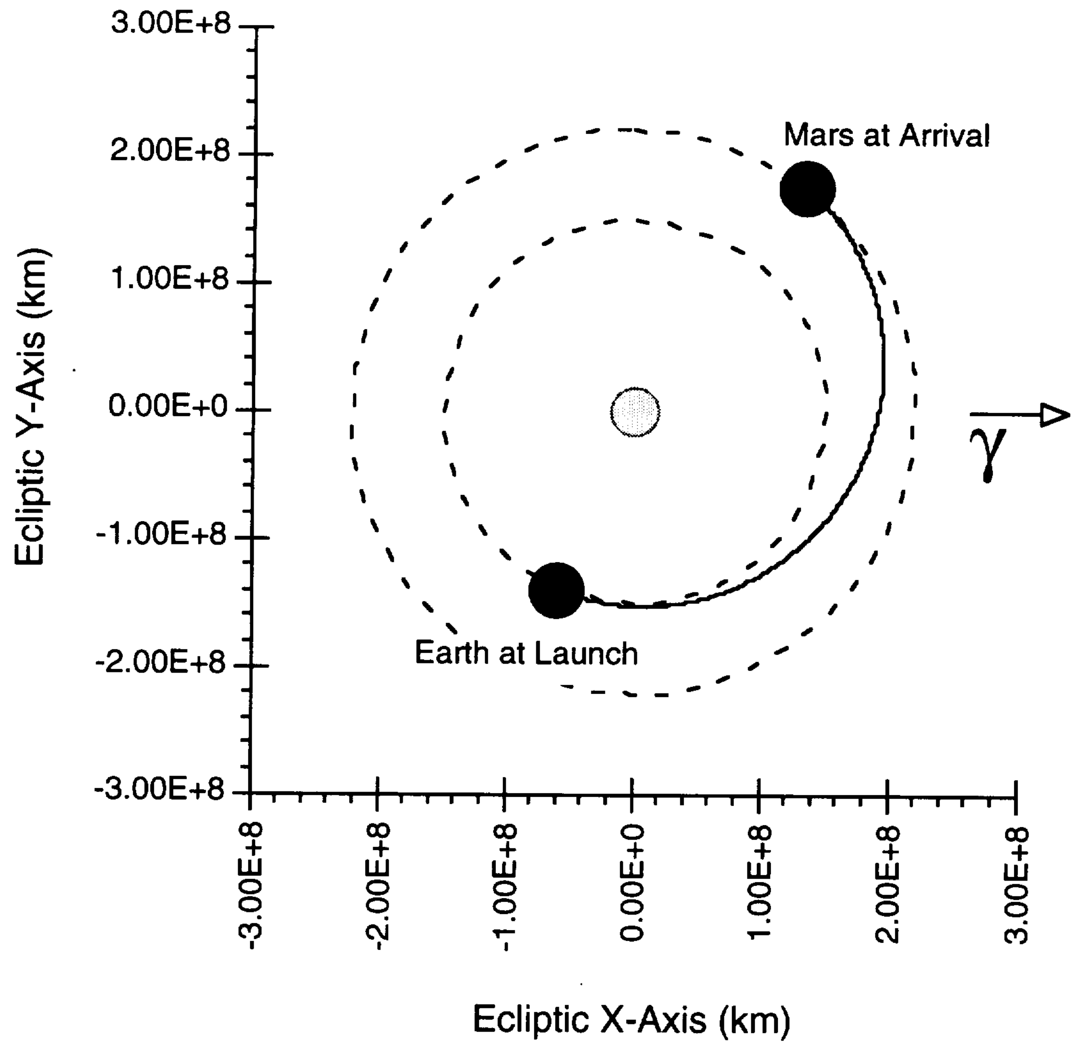

Figure 5.1.2-1: Lambert Targeted Earth-Mars Transfer Trajectory

In addition, QUICK was used to compute 3 alternate trajectories for the mission. The details of these trajectories are contained in Appendix OT-E. One of the trajectories was computed as a minimum launch C3 Type II trajectory. A Type II trajectory is one with a transfer angle (difference in true anomaly at launch and arrival) greater than 180° while a Type I trajectory is one with a transfer angle less than 180° (180° being a Hohmann transfer). The Type II trajectory computed allows a time of flight 30 days greater than the Type I trajectory but arriving at Mars within a day of the Type I arrival date.

The two other trajectories were computed for mimimum arrival C3 by using an algorithm similar to that used for computing the previous two trajectories. The minimum arrival C3 trajectories were identified for the case in which the spacecraft mass constrained the arrival ΔV's to a value lower than those obtained with the minimum launch C3. It must be noted that in general, for trajectories with similar times of flight, an increase in the launch C3 will also result in a higher arrival C3. However, minimizing the launch C3 will not, in general, minimize the arrival C3.

5.1.3 Broken Plane Trajectories

In addition to the Lambert targeted trajectories, the Orbits and Trajectory Element considered using what are know as "broken plane" trajectories for the interplanetary transfer. The broken plane is a 3 burn trajectory which carries out the plane change between Earth and Mars in 3 steps: launch ΔV , midcourse ΔV , and arrival ΔV. Broken plane trajectories are required in situations where a two burn (Lambert or Hohmann) trajectory would require very large plane change ΔV's due to the position of the departure and arrival bodies. These transfer trajectories exhibit near-polar orbital planes which result in very high values of launch C3. The broken plane trajectory can significantly reduce the departure C3 compared to a 2-burn transfer for the same departure and arrival dates. However, the need to use a broken plane trajectory can be eliminated by simply changing the arrival and departure points by a slight amount (i.e. changing the departure and arrival dates). Doing this causes the near polar transfer plane to quickly settle down into one with a much lower inclination. As a result, the added complexity and cost of performing 3-burns becomes unnecessary if one is willing to be flexible with the departure and arrival dates. In addition, the small inclination of the Martian orbit with respect to the ecliptic plane means that most Lambert targeted trajectories have a lower value of launch C3 than the equivalent broken plane trajectory.

5.2 Mars Orbit and Phobos Targeting

Project Aeneas will deliver a total of 3 spacecraft to Martian orbit. Mars-Silva (MS) 1 and 3 will be placed in LMO at an altitude of 483 km. Mars-Silva 2 will be placed at an orbit altitude equal (within several kilometers) to that of Phobos. Since Mars-Silva 1 and 2 will be launched simultaneously on a single booster they will complete most of the interplanetary trajectory attached to each other The targeting of the MS- 1 to LMO and MS-2 to Phobos orbit altitude will be carried out several days before Mars orbit insertion (MOI). The MS-1 and MS-2 spacecraft will separate and each will impart a small ΔV (several times smaller that either $\Delta V1$ or $\Delta V2$) which will target the two spacecraft to 2 different points on the arrival B-plane. The B-plane is a targeting plane perpendicular to the arrival V_∞ vector, with the intersection of this plane and the target planet's equatorial plane acting as the main reference line. The MS-3 spacecraft will be targeted in the same way as MS-1. Figure 5.2-1 illustrates the targeting setup for the Mars-Silva spacecraft.

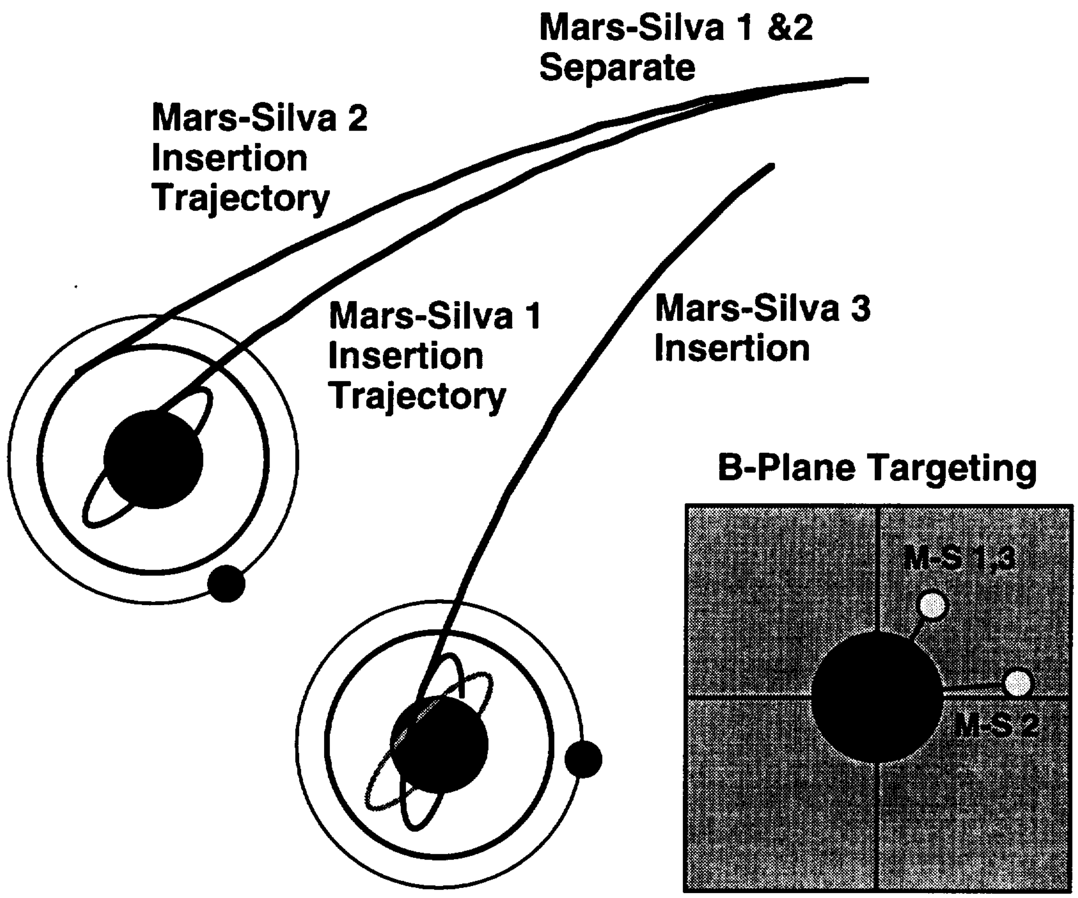

Figure 5.2-1: Diagram of Mars-Silva Spacecraft Probe Targeting

5.3 Mars Orbit

The Mars-Silva 1 and 3 spacecraft will be positioned into near circular orbits with an inclination near 60° and an altitude of about 483 km. The inclination of 60° has been chosen because of science requirements which state that regions between ±60° latitude on the surface of Mars must receive maximum coverage from the orbiters. In addition, the Science Element has determined that coverage of the poles is not required to achieve the goals of Project Aeneas.

Initially, the Orbits and Trajectory Element investigated the use of a sun-synchronous orbit for the single spacecraft concept. Sun-synchronous orbit means that the J2 induced precession of the spacecraft is equal to the observed rate of change of the Sun's longitude with respect to non-rotating coordinates fixed on the planet. An altitude can be chosen such that the orbiter is constantly in view of the sun. View of the Earth is interrupted only when the Earth is behind the Sun (a geometry known as conjunction) and when antenna attitude constraints do not allow it to point at Earth (e.g. when the Earth-Sun-Mars angle is near 90°).

A sun-synchronous orbit has some advantages over other orbit types. First, it would increase Earth-view time, an important factor in communicating with Earth. Second, this type of orbit would increase the sun exposure time. This is important to the power generating capability of the solar arrays onboard the orbiter. On the other hand, the fact that a sun-synchronous orbit is near polar places constraints on the targeting of probes to the surface of Mars. Deploying probes along a track perpendicular the orbiters footprint would require rather large ΔV's and would make the targeting more difficult. Also, constant orbit maintenance would be required to keep the sun-synchronous condition throughout the life of the mission.

The Orbits and Trajectory Element has determined that a 60° inclination orbit would satisfy the requirements of the mission as defined by the Science Element. The main factor which has driven this decision is to be able to carry out a mapping of sites between ±60 latitude within a reasonable amount of time (e.g. a couple of months). An investigation of the ground tracks of this type of orbit revealed that complete instrument coverage can be achieved within a year (a science constraint). Launching probes from a 60° orbits will reduce the separation ΔV imparted to the probes in addition to making their atmospheric entry velocities much smaller.

5.4 Phobos Targeting

The MS-2 spacecraft requires a near equatorial orbit so that it can match the orbit of Phobos. This orbit will be circular (as is Phobos') and will be slightly different in semi-major axis than the orbit of Phobos. The semi-major axis will be slightly smaller if Phobos is ahead of the spacecraft at injection. This smaller semi-major axis will allow the spacecraft to chase Phobos and eventually reach it. The spacecraft will approach Phobos at a small relative velocity which will facilitate targeting of the penetrator to the crater Stickney (the intended impact site of the probe). The semi-major axis will be slightly larger if the spacecraft is injected ahead of Phobos. This will allow Phobos to chase the spacecraft and approach it at a low relative velocity. The use of inclined (i.e. non-equatorial) and eccentric orbits have not been considered because of the limitations and constraints which they would enforce on the mission. Having an inclined orbit would limit the targeting of the penetrator to the points where the orbit of the spacecraft and the orbit of Phobos meet (i.e. the ascending and descending nodes). Given that the probe is intended to have limited guidance and control capabilities, attempting to target Phobos under such constraints would be very difficult. Similarly, an eccentric-equatorial orbit would at most provide 3 opportunities to hit Phobos. The first two at the node points, as presented in the previous case. The third opportunity would arise at either periapsis or apoapsis, depending on the orbit configuration. Again, these constraints would be to difficult to handle.

5.5 Selection of Launch System

The main focus behind the launch system selection process is the determination of whether or not a particular booster/upper-stage combination can deliver the required C3 for the interplanetary transfer.

The JPL's Advanced Projects Group (Mission Design Section) has provided Argos Space Endeavours with the required reference material [OT 2] to generate C3 polynomials which plot injected mass as a function of C3 for a variety of booster/upper-stage combinations. Each combination was considered and the systems with the highest injection mass were selected as possible launch systems for Project Aeneas. Figure 5.5-1 contains injected mass vs. C3 curves for the Proton class boosters.

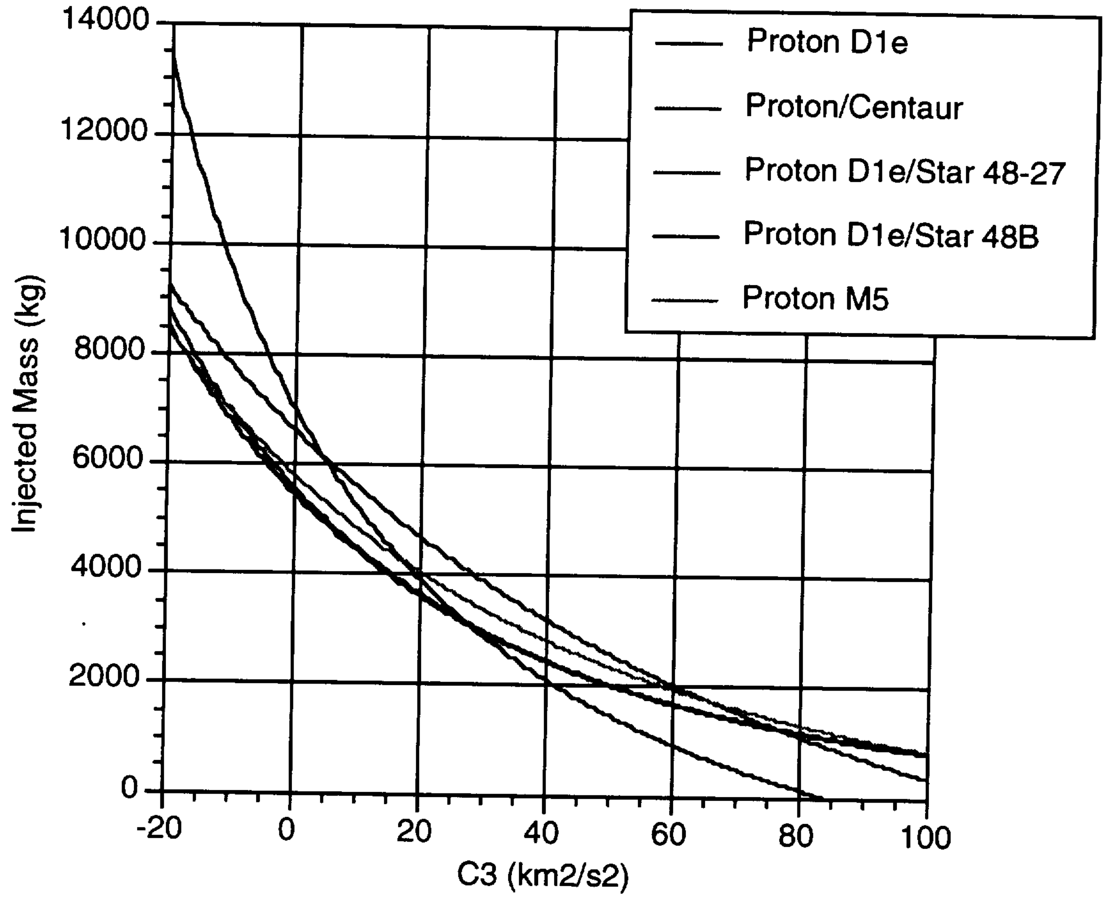

Figure 5.5-1: Proton Class Vehicle Performance

The selection of the booster/upper-stage is also dependent on the dimensions of the payload faring which each launch system can accommodate. Based on the launch C3/injected mass constraint and the volume constraint of the payload fairing, three booster/upperstage systems were considered:

<u>3-Launch Configuration</u>

Atlas II/Centaur (2 burns)

C3L Mass = 1700 kg

Cost Per Launch ~ $80 million

2-Launch Configuration

Proton/D-1e

C3L Mass = 5400 kg

Cost Per Launch ~ $40 million

1-Launch Configuration

Titan IV (SRM)/Centaur

C3L Mass = 6100 kg

Cost Per Launch > $400 million

After carefully considering the launch options available, Argos Space Endeavours has chosen the Proton/D-1e as the launch system for Project Aeneas due to its relatively low cost.

See Appendix OT-F for cost vs. payload mass comparisons between various boosters.

6.0 Management Report

6.1 Argos Space Endeavours Organization

Argos Space Endeavours utilized a small team of engineers for the preliminary design effort since a limited number of engineers will make communication during the design process simpler. Appendix M-A shows an organizational chart for the team. Seven engineers shared the responsibilities for all technical and administrative work necessary for the timely completion of the Aeneas project. Since the number of engineers assigned to the project was limited, all members were required to serve in several elements during the course of the design effort. For this reason Argos engineers with expertise in many areas were selected for completion of the project; however, the project required team members to develop new expertise in areas with which they were not familiar.

The organization is divided into upper management and the design elements. The upper management consists of the Chief Executive Officer (CEO), the Chief Engineer (CE), and the Administrative Officer (AO). The CEO is the single point of contact for all interactions with the contract monitor. It is the CEO's responsibility to create and maintain the project schedule and assign manpower for all planned activities. Appendix M-B contains the Aeneas project schedule. The CEO makes all final recommendations for the project design once input is received from the CE and the element leaders. The CE is responsible for ensuring that all elements complete their assignments on schedule. The CE also provides technical guidance for the element leaders and serves as the lead integrator for the project. It is the CE's responsibility to monitor all technical aspects of the project and verify that the system designs interact in a manner such as to meet all project goals. The AO is responsible for tracking man-hours and maintaining the project notebook. Each team member must submit a timecard each week documenting the progress made and number of hours worked on project activities. All memos and communications internal and external to Argos Space Endeavours are documented by team members and indexed by the AO. All element products such as system designs and calculations are indexed by the AO as well.

The design elements are responsible for all technical aspects of the project. Each design element has an element lead which reports directly to the CE. Design problems flagged by element members can be communicated to the CE through the element lead. Element members may also use weekly element status meetings with the CEO to discuss design flaws or issues.

6.2 Manpower Utilization

Based on manpower reports generated by the AO, the CEO was able to determine whether element manpower was under or over utilized. The AO produced the manpower reports based on documented activities found in the weekly timecards. The timecards will documented the amount of time spent on each activity for the week. Figures 6.2-1 and 6.2-2 summarize the manpower utilization for the Project Aeneas design effort. Element leads also communicated manpower needs through the CE. Since the number of engineers assigned to the Aeneas project is small, element members and upper management worked on more than one design element.

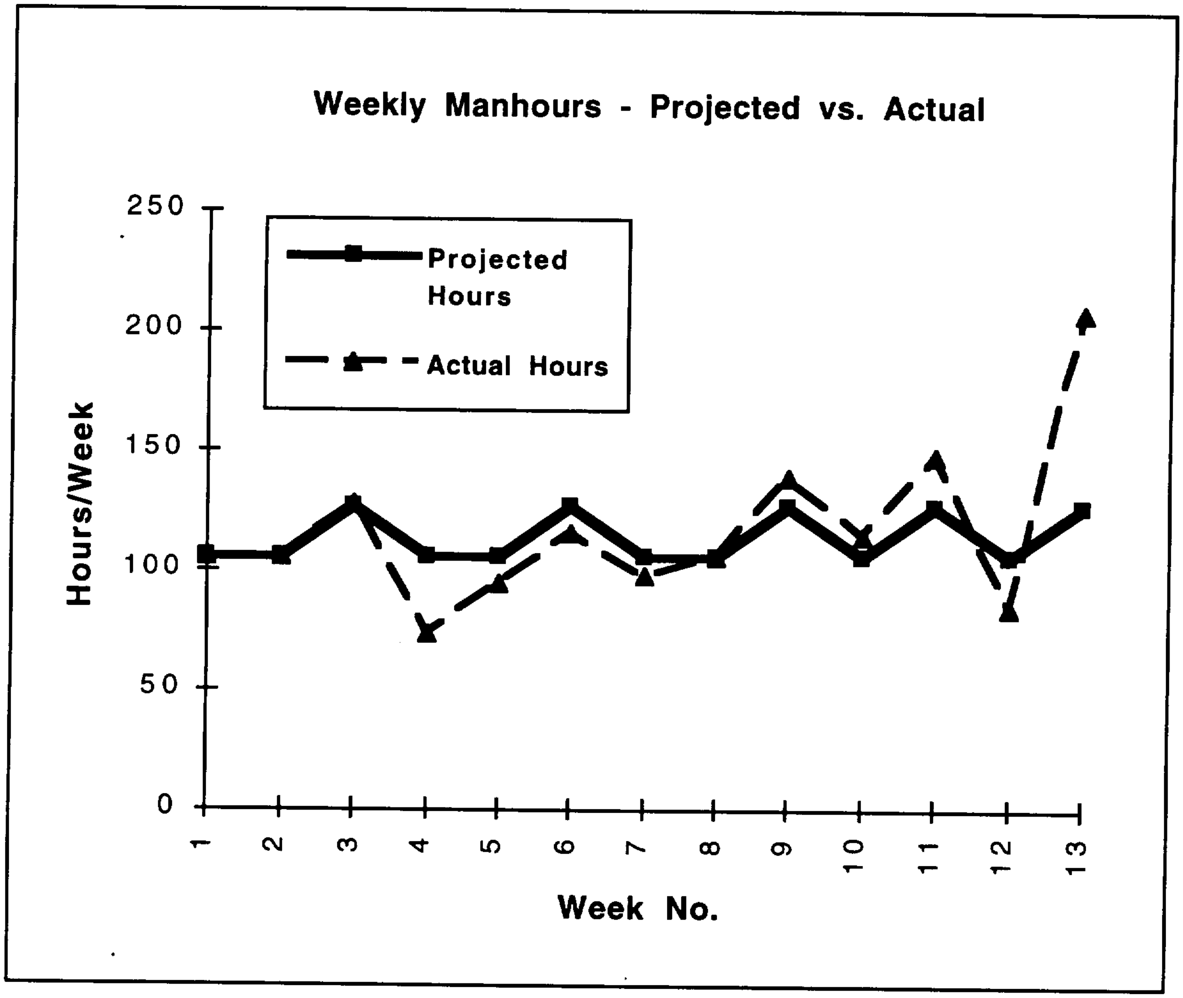

Figure 6.2-1: Weekly Manhours-Projected vs. Actual

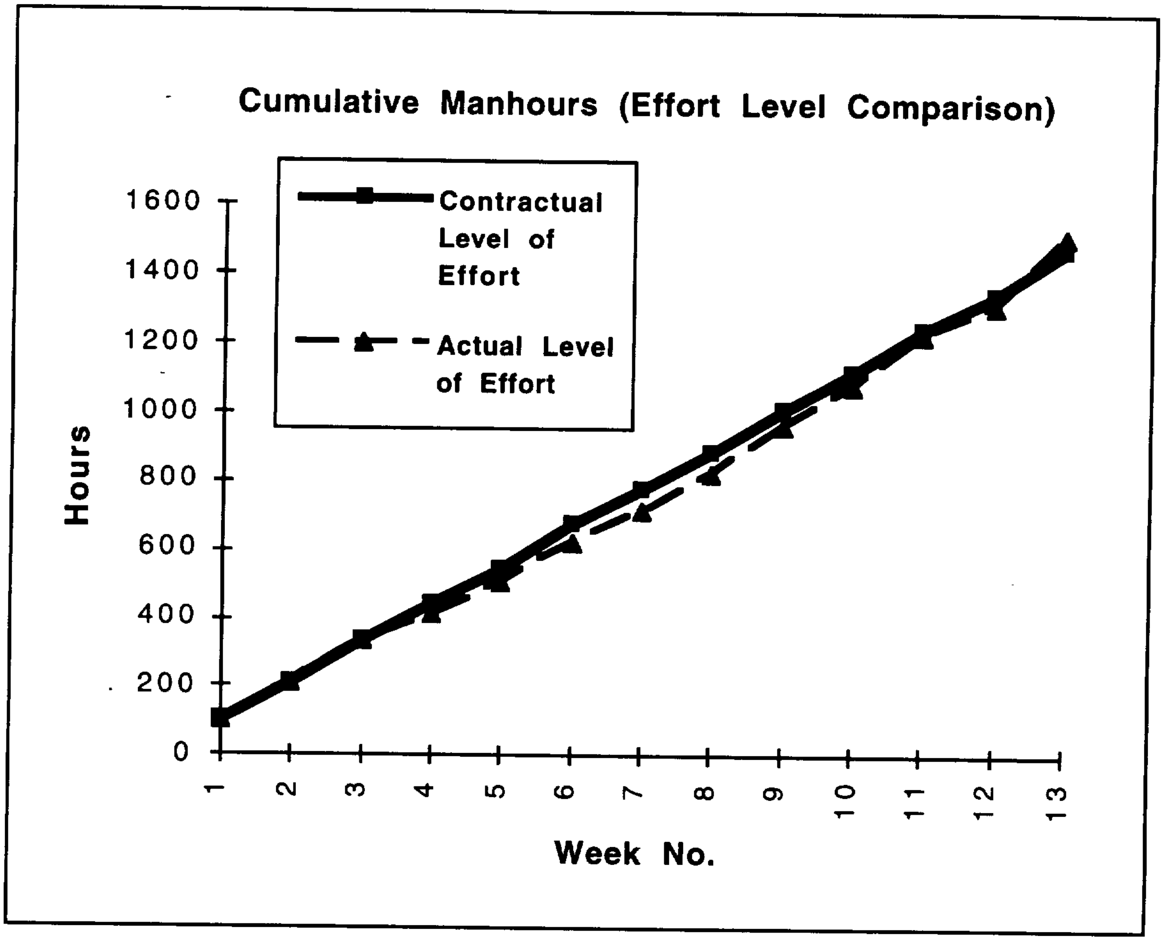

Figure 6.2-2: Cumulative Manhours-Projected vs. Actual

Since the Aeneas design team is small, project integration was simplified by assigning engineers from one element-to work on another over utilized element for a short period of time. In this way members from both elements gained an understanding of design issues from different perspectives. In short, the level of awareness of other elements activities was increased.

6.3 Personnel Costs

The following is a list of Argos Space Endeavours personnel currently working on the Aeneas Project and their salary figures as documented in the ASE proposal.

Name	Title	Hourly Rate ($)
Kerr	CEO / Engineer	45
Defosse	Chief Engineer	42
Ho	Admin. Officer / Engineer	35
McCourt	Engineer / Artist	35
Smith	Engineer / Artist	35
Barriga	Engineer	32
Davis	Engineer	32

Each team member submitted a timecard on Friday of each week. The timecard values were documented and final project costs were computed. For the ASE proposal, the following project cost estimation scheme was used. On a normal work week, the estimated number of working hours per person was 15. However, on a week with a presentation, the estimated number of working hours was increased to 18, or by 20%. For the 14 weeks of the project, 9 are considered normal weeks and 5 are considered presentation weeks. The 14-week total personnel cost estimate is compared to the timecard data below.

Name	14-Week Total Proposal Estimate ($)	14 Week Total Timecard Data ($)
Kerr	10,125	11,970
Defosse	9,450	10,626
Ho	7,875	8,417.5
McCourt	7,875	6,317.5
Smith	7,875	6,965
Barriga	7,200	5,280
Davis	7,200	6,720
Subtotal	**57,600**	**56,296**

6.4 Computer Costs

The projected costs of computer time and supplies are as follows. These computer cost estimates did not change during the course of the design effort.

Computer	Cost/Week Per Employee	Total Usage Cost ($)	Supply Cost ($)	Total Cost ($)
Macintosh	20	1,960	50	2,010
IBM PC	5	490	0	490
DEC Alpha Workstations	10	980	0	980
UNIX Main Frames	5	490	5	495
Subtotal	40	3,920	55	**3,975**

6.5 Material and Miscellaneous Costs

Materials and other miscellaneous costs are as follows. These estimates also did not change during the course of the semester.

Cost Item	Costs ($)
Photocopies @ $0.08 ea.	50
View Graphs @ $0.50 ea.	75
Physical Design Model	70
Project Poster	60
Long Distance Telephone Calls	125
Planned Trips	150
Miscellaneous Supplies	25
Subtotal	**555**

6.6 Total Project Cost

The total final cost for the Aeneas Project preliminary design effort including personnel, computer and other costs is therefore **$60,826** as compared to the proposed $62,130. Since the design effort was completed under budget and on time, Argos Space Endeavours should be awarded an additional 15% as described in the RFP. Therefore, the final total PDR effort cost plus bonuses is **$69,950**.

7.0 Recommendations

- Develop the ISRU vehicle in detail (possibly a project that should be handled by ASE 363Q)
- Develop the Penetrator structural design (possibly a project that should be handled by ASE 363Q)
- Carry out a more in-depth analysis of the trajectory issues. In particular the targeting of the spacecraft into Mars orbits and the final form of the Mars orbits themselves.
- Carry out a more in-depth analysis and design on the spacecraft and its subsystems. The work carried out by ASE is preliminary and is only intended to provide an overall spacecraft design which would be suitable for a mission like Project Aeneas.
- Investigate the targeting issues involved in delivering probes to the surface of Mars and Phobos. In particular, develop a model for the thermal environment that the probes will encounter upon entering the Martian atmosphere. Also, develop guidance and control systems which will ensure that the probes are delivered accurately.
- Carry out a detailed study on how the orbiters will map the surface of Mars in preparation for the deployment of probes. Generate groundtracks and figure out how (in terms of orbit design) to maximize the coverage of interested locations on the surface.
- Consider adding studies of micro meteoroid impacts on the Martian surface and radiation levels during the cruise phase to Mars.

8.0 References

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Appendix S-A: Instrument Characteristics

Gamma-Ray Spectrometer (GRS) [S4;3-5]

Instrument Builders: Goddard Space Flight Center; Martin Marietta Astronautics Group

Experiment Objectives

1. Determine the elemental composition of the surface of Mars with a spatial resolution of a few hundred kilometers through measurements of incident gamma-rays and albedo neutrons (H, O, Na, Mg, Al, Si, S, Cl, K, Ca, Ti, Mn, Fe, Th, U, and C)
2. Determine the hydrogen depth dependence in the top tens of centimeters.
3. Determine the atmospheric column density.
4. Determine the seasonal variation in polar cap thickness.
5. Determine the arrival time and spectrum of gamma-ray bursts.

The GRS instrument is designed to carry out its objectives by measuring the intensity of gamma-ray lines, characteristic of each element, that emerge from the planetary surface.

General Characteristics

Power	14.0 W
Mass	23.2 kg
Data Rate	665 b/s

High Resolution Camera (HRC) [S4;12-15]

Experiment Objectives:

1. Obtain global synoptic views of the Martian atmosphere and surface to study meteorological, climatological, and related surface changes.
2. Monitor surface and atmosphere features at moderate resolution for changes on time scales of hours, days, weeks, months, and years.
3. Systematically examine local areas at extremely high spatial resolution in order to quantify surface/atmosphere interactions and geological processes.

General Characteristics

Power (approximate)	7.5W average, 25.7 W peak
Mass (approximate)	21 kg
Data Rate (approximate)	1, 3, 9, 11 kb/s recorded, 30-40 kb/s real time

Thermal Emission Spectrometer (TES) [S4;24-26]

Instrument Builder: Santa Barbara Research Center

Experiment Objectives:

1. Determine and map the composition of surface minerals, rocks, and ices.
2. Study the composition, particles size, and spatial and temporal distribution of atmospheric dust.
3. Locate water-ice and carbon-dioxide condensate clouds and determine their temperature, height, and condensate abundance.
4. Study the growth, retreat, and total energy balance of the polar cap deposits.

5. Measure the thermophysical properties of the Martian surface (thermal inertia, albedo) used to derive surface particle size and rock abundance.

General Characteristics

Power	13.2 W
Mass	14.1 kg
Data Rate	688 and 1664 b/s recorded and 4992 b/s real time

Appendix S-B: Program Cylinder

```
      program cylinder


      implicit double precision(a-h,o-z)
c
c     *** Variable Dictionary ***
c
c     alt - altitude (km)
c     alt_max - final (highest) altitude (km)
c     rho - density (kg/km3)
c     rho_position - point where density is taken (km)
c     speeds - speed of sound (km/s ?)
c     radius - radius of cylinder (km)
c     length - length of cylinder (km)
c     pi - pi
c     volume - volume of the cylinder (km^3)
c     step_mass - mass at one step (kg)
c     total_mass - the total mass (kg)
c     comass - mass of CO2 in the sample (kg)
c     methmass - mass of CH4 which can be ideally produced by the CO2
c     limit - the mass required of CO2
c     idens - unknown function for cospar routine
c     dscale - unknown function for cospar routine
c     step - step counter
c
```

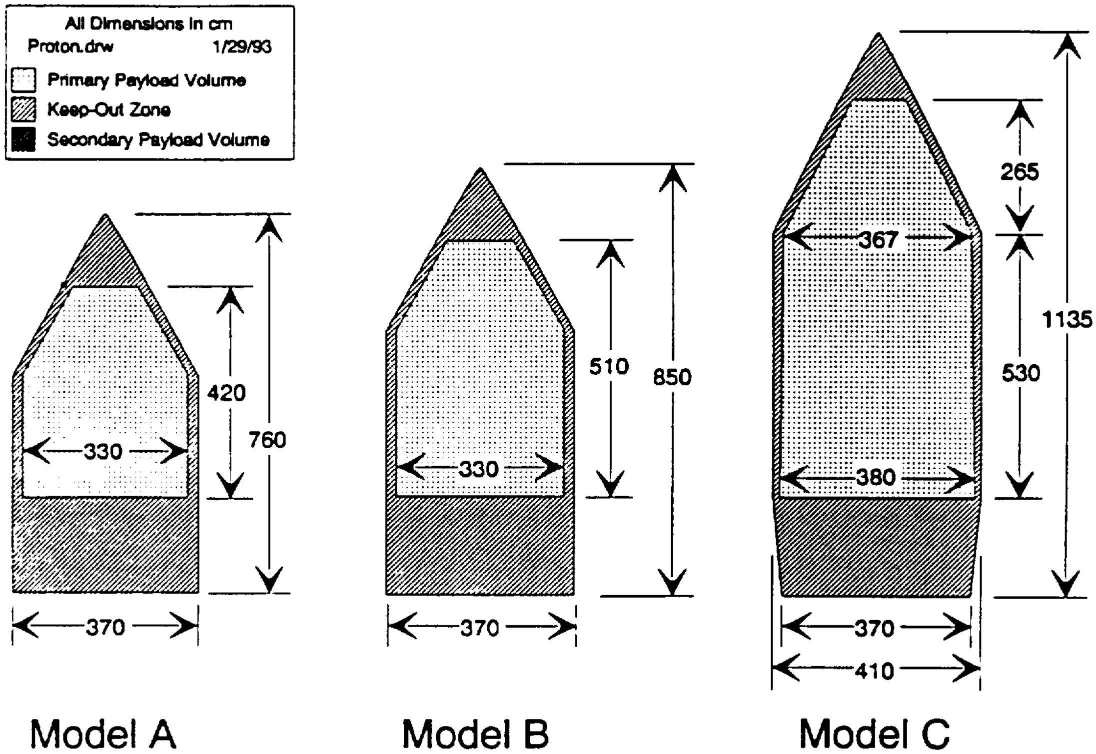
```
      double precision alt, alt_init, rho, rho_position, speeds
      double precision radius, length, pi, volume
      double precision step_mass, total_mass, comass, methmass, limit
      integer idens, dscale, step
c
c     initialize variables
c
      idens = 4
      dscale = 1
      alt_max = 1d1
      alt = 15d0
      radius =  20d-5
      length = 1d-2
      rho_position = alt + length / 2
      pi = dacos(-1d0)
      total_mass = 0d0
      comass = 0d0
      methmass = 0d0
      limit = 1d0
c
c     calculate the volume of the cylinder
c
      volume = radius**2*pi*length
c
c     write the labels
c
      write(*,*) "Altitude  Rho_Alt  StepMass CO2Mass CH4Mass"
```

```
c
c   Start the while loop
c
      do while (methmass .le. limit)
c
c   get the density, speed of sound
c
        call cospar(idens,dscale,rho_position,rho,speeds)
c
c   calculate the mass present in this step
c
      step_mass = rho*volume
c
c   add the mass of this step to the total mass
c
      total_mass = total_mass + step_mass
      comass = 0.6d0 * 0.97d0 * total_mass
      methmass = comass / 2.7433d0
c
c   write the results
c
      write(*,1000) alt, rho_position, step_mass, comass, methmass
c
c   move to the next altitude
c
      alt = alt - length
      rho_position = alt + length / 2
```

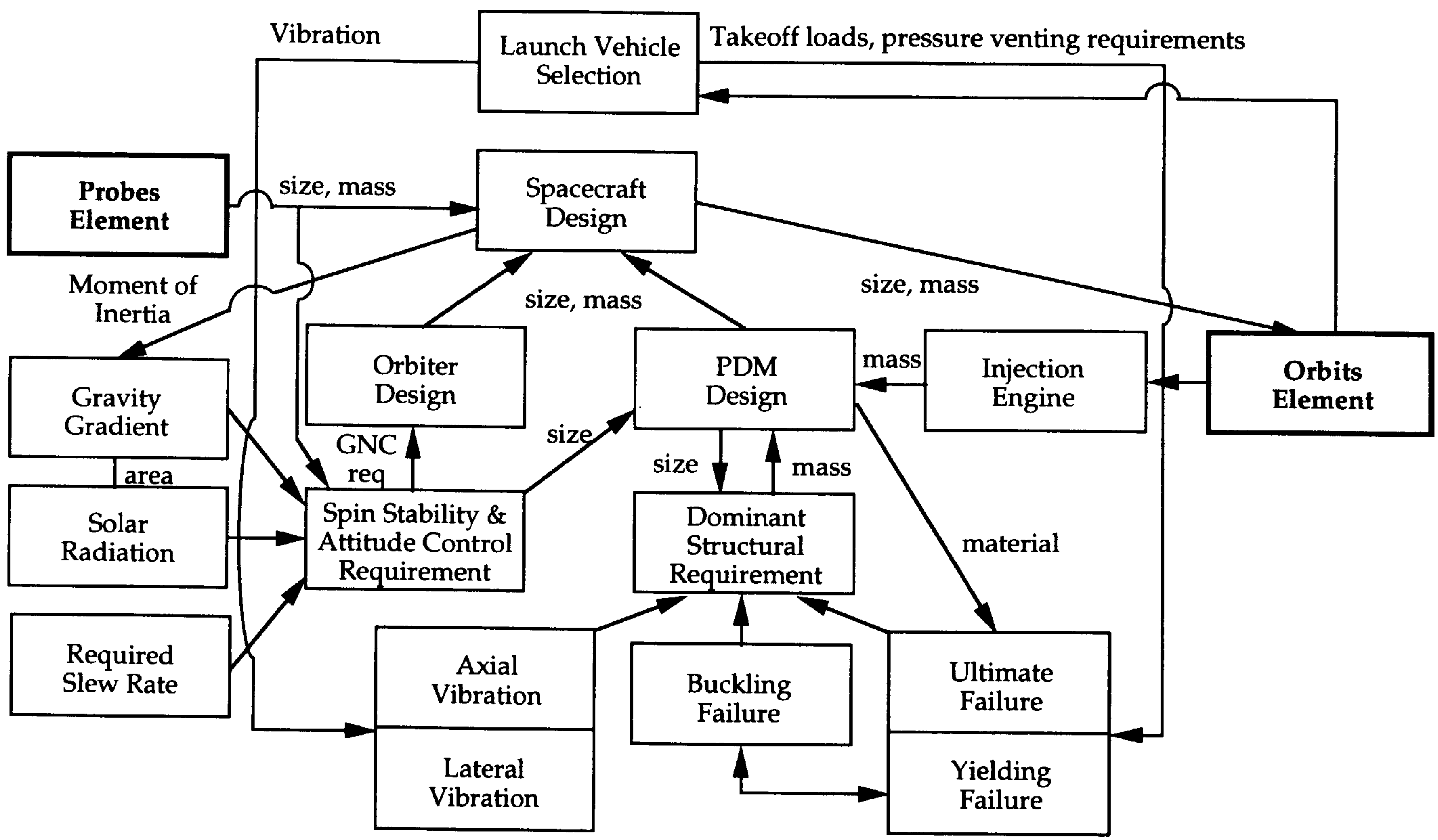
```
c
c   end the do loop
c
    end do
c
c   format statement
c
 1000 format (1x, 6(e9.4, 2x))
    end
```

Appendix Sp-A: Proton Launch Vehicle Configurations

PROTON PLF CONFIGURATIONS



Appendix Sp-B: Mars-Silva Design Functional Flow Diagram



Vibration
Launch Vehicle Selection
Takeoff loads, pressure venting requirements
Probes Element
size, mass
Spacecraft Design
Moment of Inertia
size, mass
size, mass
Orbiter Design
PDM Design
mass
Injection Engine
Orbits Element
Gravity Gradient
GNC req
size
size
mass
area
Spin Stability & Attitude Control Requirement
Dominant Structural Requirement
material
Solar Radiation
Required Slew Rate
Axial Vibration
Buckling Failure
Ultimate Failure
Lateral Vibration
Yielding Failure

Appendix P-A: Penetrator Data

Table PA-1: Penetrator Performance in Hard & Soft Soil*

	Hard Soil	Soft Soil
Initial Velocity	240 m/s	240 m/s
Impact Velocity	80 m/s	80 m/s
Total Penetration	1.65 m	4.08 m
Antenna Height	0.48 m	(-)0.12 m
Fore Maximum Deceleration	2234 g	787 g
Aft Maximum Deceleration	12671 g	4516 g

* [P 1;49, 50]

Table PA-2: Penetrator Stresses and Safety*

Section	Maximum Stress (MPa)*	Allowable Stress (GPa)*	Margin of Safety**	Critical Margin of Safety***
Fore	245	13.6	54	0.23
Aft	253	2080	8219	0.22

*[P1;54]

**[P1;55]

***[P1;57]

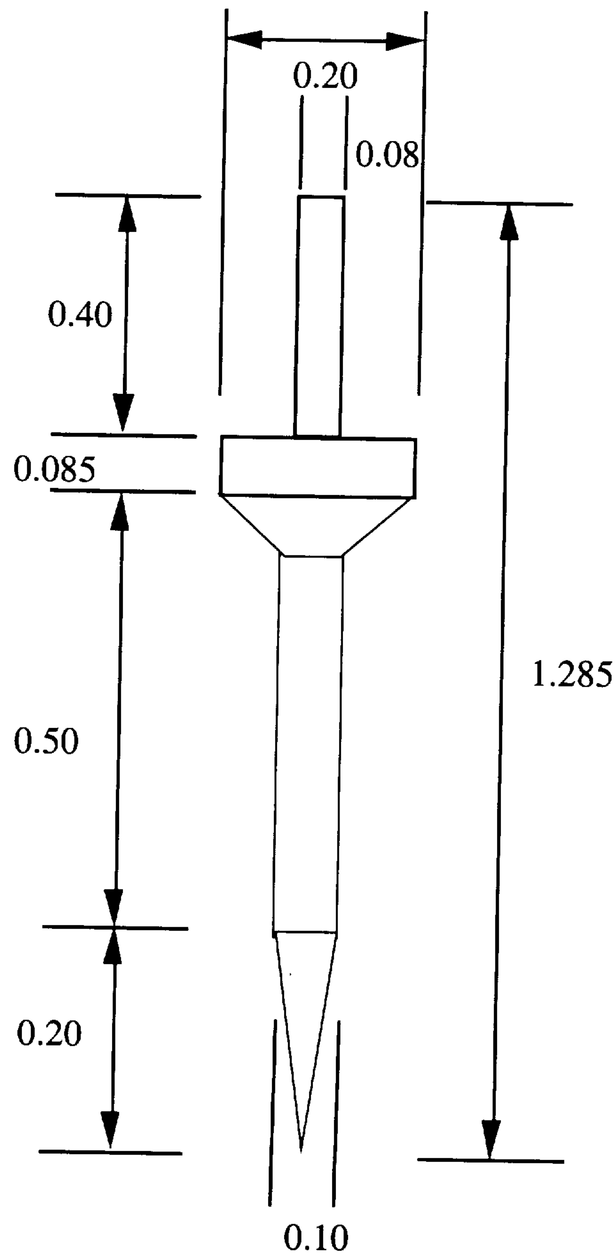


Figure P A-1: Penetrator Dimensions (meters)

Appendix P-B: Micro-rover Data

Table PB-1: Micro Rover Accommodations*

Computer	80C85 Dual Speed CPU
work mode	200 Kips with 512 Bytes of RAM
Camera	Kodak KAI-0370 CCD Array
transmission rate	8 KBytes/sec
Communications Link	Motorola RF Modem
range	1.2 mi. at 9600 baud
antennas	39.4 in. whips
Material Analysis Technique	Alpha-Proton-X-Ray Spectrometer

* [P2; 4]

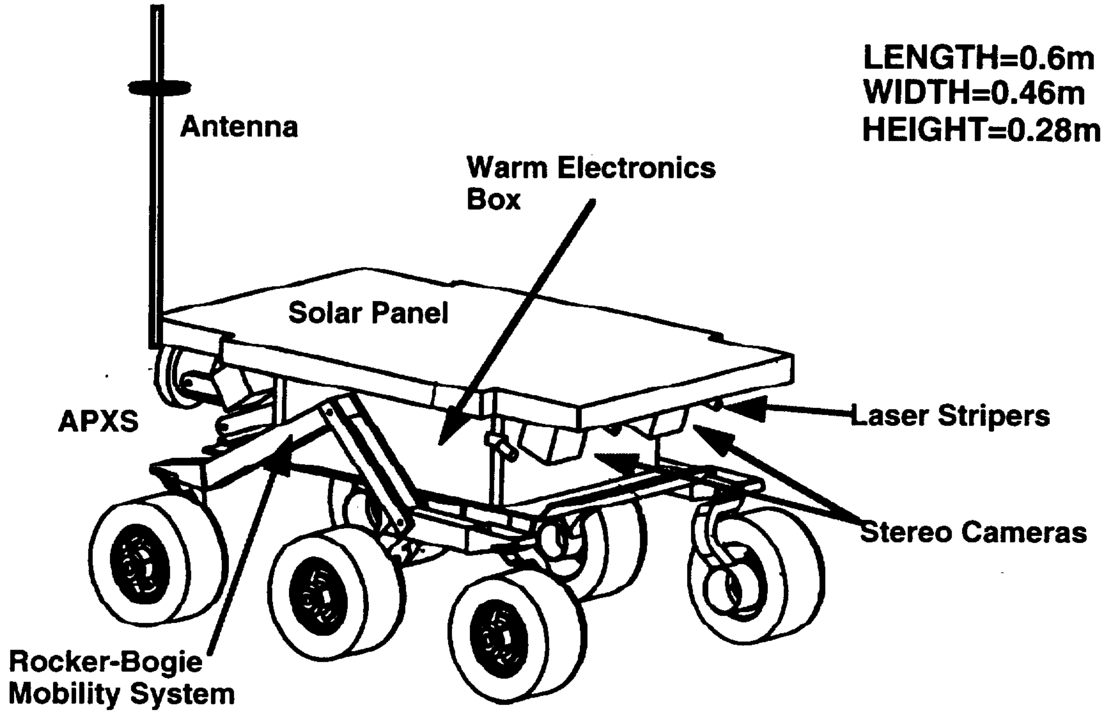


Figure P B-1: Micro-rover

Appendix P-C: CRAF Data

Table P C-1: CRAF Data

Length	Maximum Load	Propulsion	Velocity Delta
1.3 m	600 g	22 N	60 m/s

Appendix P-D: Probe Combination Trade Study

Table P-D-1: Probe Combination Options

Cases 1 through 4

Case 1

	QUANTITY	MASS (KG)	COST (MIL$)
	MARS-SILVA 1		
CRAF	1	100	30
PENETRATOR	3	83.1	22.5
ISRU	1	30	10
ORBITER	1	250	50
SUBTOTAL		463.1	112.5
	MARS-SILVA 2		
ROVER	1	125	75
CANISTER	2	140	60
ORBITER	1	250	50
SUBTOTAL		515	185
	MARS-SILVA 3		
PENETRATOR	3	83.1	22.5
ISRU	1	30	10
CANISTER	2	140	60
ORBITER	1	250	50
SUBTOTAL		503.1	142.5
TOTAL		**1481.2**	**440**

This spacecraft configuration was not chosen because of the high cost. Furthermore, the rover was not being considered desirable since it uses RTG. Lastly, there was only one CRAF probe and that it is mixed with Mars probes on Mars-Silva 1.

Case 2

	Quantity	Mass (kg)	Cost (M$)
	Mars-Silva 1		
Rover	1	125	75
Penetrator	3	83.1	22.5
ISRU	1	30	10
Orbiter	1	250	50
Subtotal		488.1	157.5
	Mars-Silva 2		
CRAF	1	75	30
Subtotal		75	30
	Mars-Silva 3		
Penetrator	3	83.1	22.5
ISRU	1	30	10
Canister	2	140	60
Orbiter	1	250	50
Subtotal		503.1	142.5
Total		**1066.2**	**330**

This spacecraft configuration was not chosen because it consists of only 1 CRAF probe. Furthermore, it also include a rover that uses RTG. Lastly, all canisters are located on the third spacecraft. Therefore if Mars-Silva 3 fails, all canisters are lost.

Case 3

	Quantity	Mass (kg)	Cost (M$)
	Mars-Silva 1		
Penetrator	7	193.9	52.5
ISRU	1	30	10
Orbiter	1	250	50
Subtotal		473.9	112.5
	Mars-Silva 2		
CRAF	1	75	30
Subtotal		75	30
	Mars-Silva 3		
Penetrator	3	83.1	22.5
ISRU	1	30	10
Canister	2	140	60
Orbiter	1	250	50
Subtotal		503.1	142.5
Total		**1052**	**285**

This spacecraft configuration was not chosen because it consists of only 1 CRAF. Also it has only 2 canisters on Mars-Silva 3. Therefore, it Mars-Silva 3 fails, all canisters are lost.

Case 4

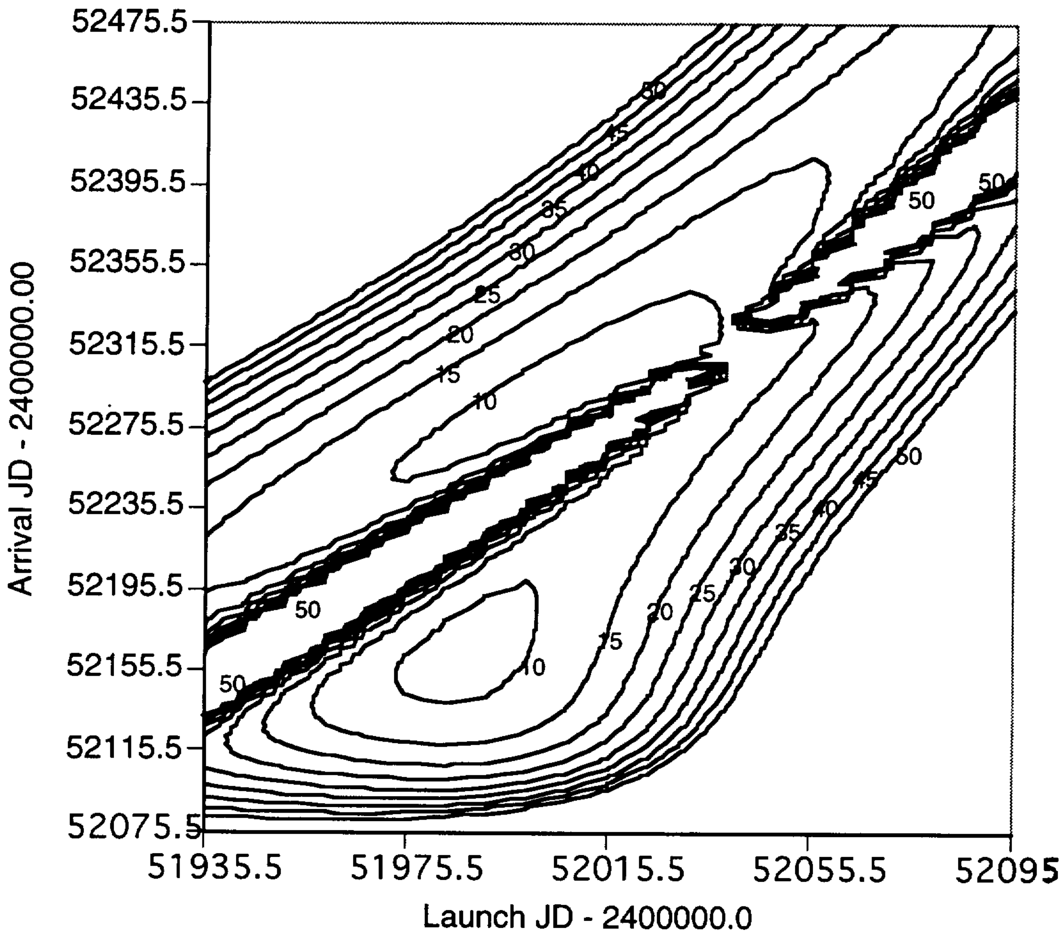
	Quantity	Mass(kg)	Cost (M$)
	Mars-Silva 1		
Penetrator	7	193.9	52.5
ISRU	1	30	10
Orbiter	1	250	50
Subtotal		473.9	112.5
	Mars-Silva 2		
CRAF	1	75	30
Subtotal		75	30
	Mars-Silva 3		
Penetrator	3	83.1	22.5
ISRU	1	30	10
Rover	1	125	75
Orbiter	1	250	50
Subtotal		488.1	157.5
TOTAL		**1037**	**300**

This option was rejected because it has no canister and only 1 CRAF. Furthermore, the rover uses RTG.

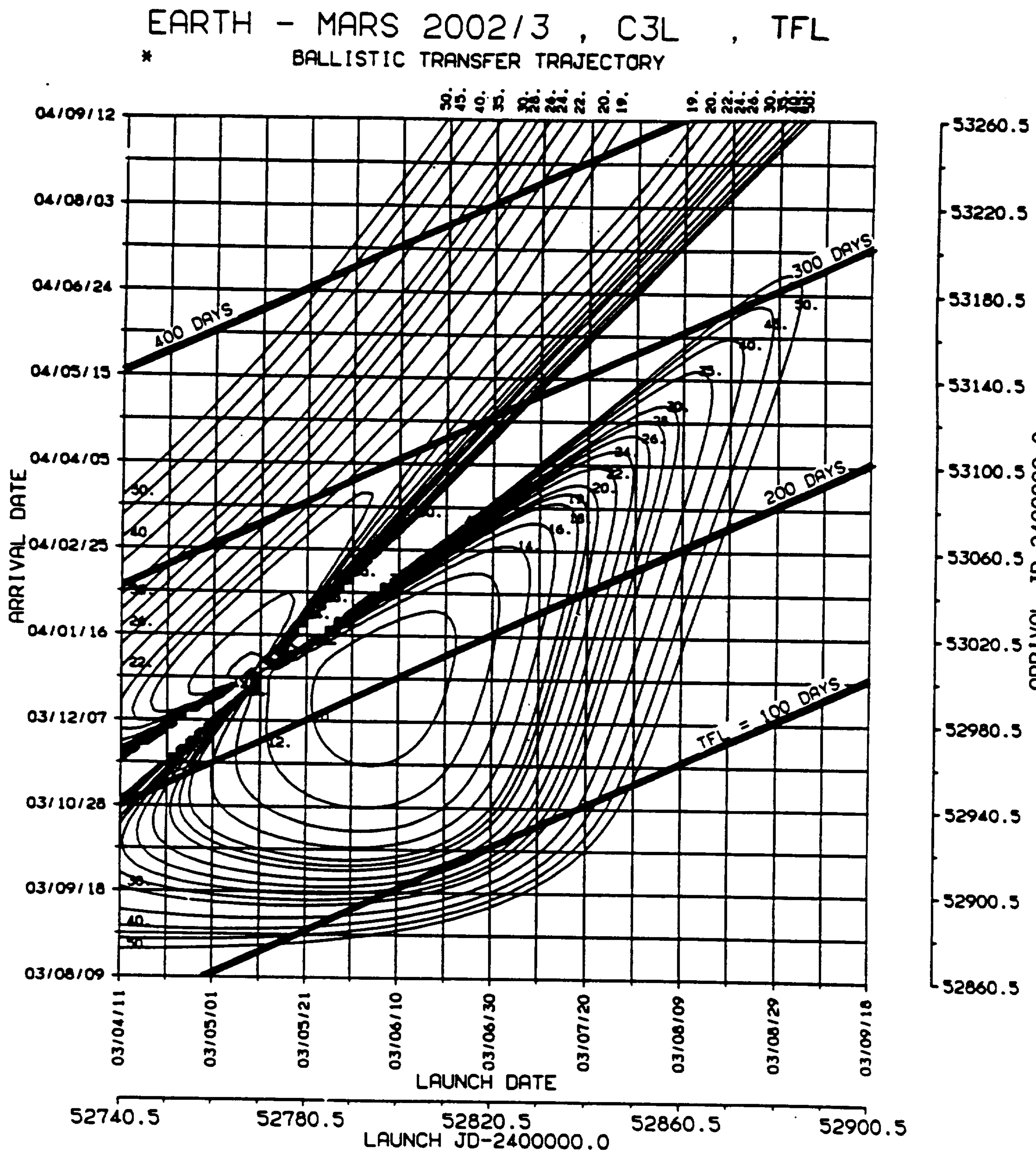
Appendix OT-A: Sample TK Solver Model For ΔV Computations

St	Input	Name	Output	Unit	Comment
	42830	gmp		km3/s2	GM of Mars
	398600.48	gme		km3/s2	GM of Earth
	6378.145	re		km	Radius of Earth
	3397.2	rp		km	Radius of Mars
	8.8095	c3l		km2/s2	Launch C3
		vinfl	2.9680802	km/s	Launch V-Infinity
	150	leo		km	LEO Altitude
		vcleo	7.8140109	km/s	LEO Circular Vel
		vleo	11.442335	km/s	LEO Injection Vel
		dvleo	3.6283242	km/s	Injection ΔV
	7.3163	c3a		km2/s2	Arrival C3
		vinfa	2.704866	km/s	Arrival V-Infinity
L	483	h		km	Mars Orbit Alt
L		rc	3880.2	km	Mars Orbit Radius
		vc	3.3223622	km/s	Mars Orbit Velocity
		v	5.4214833	km/s	Insertion Velocity
L		dv	2.0991211	km/s	Insertion ΔV
		dvtot	5.7274453	km/s	Total ΔV
	9.81	g		m/s2	Acc. of Gravity
	350	Isp		sec	Specific Impulse of
Propellant					
	750	mf		kg	S/C Dry Mass
L		mo	1382.2087	kg	S/C Wet Mass
L		mfuel	632.20873	kg	Fuel Mass
		mindv	1.9602193	km/s	Min ΔV
		rcmindv	11197.2	km	Min ΔV Orbit Rad
		hmindv	7800	km	Min ΔV Orbit Alt
		momin	1327.4076	kg	Min S/C Wet Mass

Appendix OT-B: C3 Contours vs Arrival and Departure Dates for the 2001 & 2002/3 Launch Opportunities



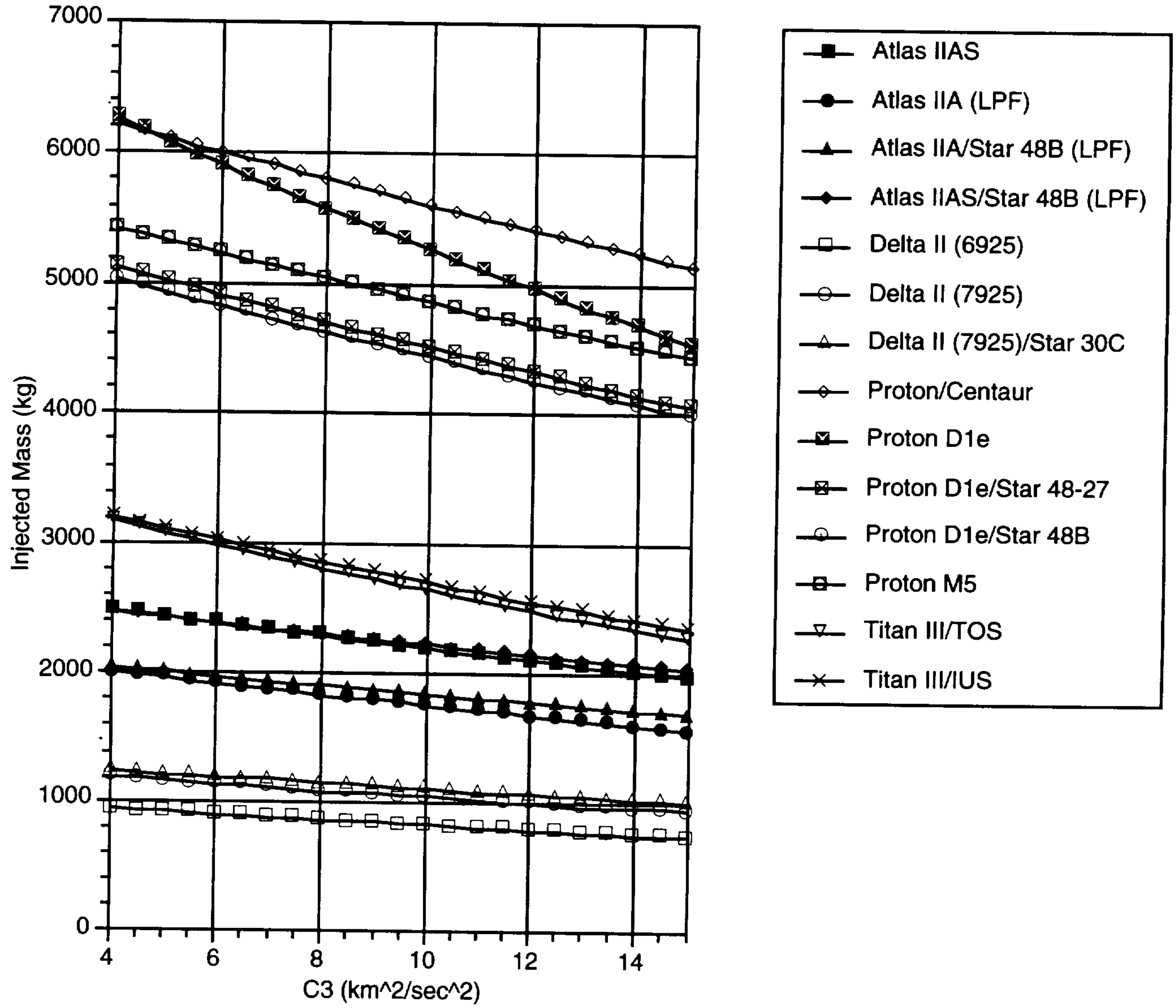
Plot of C3 vs. Arrival/Departure Dates for 2001 Launch Opportunity



EARTH - MARS 2002/3 , C3L , TFL
*
BALLISTIC TRANSFER TRAJECTORY
400 DAYS
300 DAYS
200 DAYS
TFL = 100 DAYS
ARRIVAL DATE
04/09/12
04/08/03
04/06/24
04/05/15
04/04/05
04/02/25
04/01/16
03/12/07
03/10/28
03/09/18
03/08/09
ARRIVAL JD-2400000.0
53260.5
53220.5
53180.5
53140.5
53100.5
53060.5
53020.5
52980.5
52940.5
52900.5
52860.5
LAUNCH DATE
03/04/11
03/05/01
03/05/21
03/06/10
03/06/30
03/07/20
03/08/09
03/08/29
03/09/18
LAUNCH JD-2400000.0
52740.5
52780.5
52820.5
52860.5
52900.5

113 A

Appendix OT-C: Injected Mass versus C3 for Various Booster/Upperstage Combinations



Appendix OT-D: QUICK Trajectory Optimization Input File

```
double=1

ips=3,4                       @ Vector of planet IP's (Earth=3, Mars=4)

@ input data for min C3L trajectories (trajectories #1, #2)
@ note that these dates are guesses for optimum traj performance

jdl1=date(20030607.0)         @ Launch Date, C3L min, type I
jda1=date(20031225.0)         @ Arrival Date, C3L min, type I
gjd1=jdl1,jda1                @ Vector of launch/arrival JD's

jdl2=date(20030510.0)         @ Launch Date, C3L min, type II
jda2=date(20031229.0)         @ Arrival Date, C3L min, type II
gjd2=jdl2,jda2                @ Vector of launch/arrival JD's

@ input data for min C3A trajectories
@ note that these are initial guesses

jdl3=date(20030613.0)         @ Launch Date, C3A min, type I
jda3=date(20031231.0)         @ Arrival Date, C3A min, type I
gjd3=jdl3,jda3                @ Vector of launch/arrival JD's

jdl4=date(20030510.0)         @ Launch Date, C3A min, type I
jda4=date(20031230.0)         @ Arrival Date, C3A min, type II
gjd4=jdl4,jda4                @ Vector of launch/arrival JD's

@ compute traj #1 parameters

cbodyn(jdl1,0,0)              @ Sun is central body
jd=c3min(ips,gjd1,0,1)        @ Find min launch C3 orbit

tof=(jd(2)-jd(1))*spd         @ TOF for transfer
ev=plvel(jd(1),3)             @ Earth velocity on optimal launch date
mv=plvel(jd(2),4)             @ Mars velocity on optimal arrival date

vinfl1=absv(orbvel(0)-ev)     @ Compute launch v-infinity
c3l1=vinfl1**2                @ Compute launch C3

vinfa1=absv(orbvel(tof)-mv)   @ Compute arrival v-infinity
c3a1=vinfa1**2                @ Compute C3

@ compute traj #2 parameters
```

```
cbodyn(jdl2,0,0)                @ Sun is central body
jd=c3min(ips,gjd2,0,1)          @ Find min launch C3 orbit

tof=(jd(2)-jd(1))*spd           @ TOF for transfer
ev=plvel(jd(1),3)               @ Earth velocity on optimal launch date
mv=plvel(jd(2),4)               @ Mars velocity on optimal arrival date

vinfl2=absv(orbvel(0)-ev)       @ Compute launch v-infinity
c3l2=vinfl2**2                  @ Compute launch C3

vinfa2=absv(orbvel(tof)-mv)     @ Compute arrival v-infinity
c3a2=vinfa2**2                  @ Compute arrival C3

@ compute traj #3 parameters

cbodyn(jdl3,0,0)                @ Sun is central body
jd=c3min(ips,gjd3,0,2)          @ Find min arrival C3 orbit

tof=(jd(2)-jd(1))*spd           @ TOF for transfer
ev=plvel(jd(1),3)               @ Earth velocity on optimal launch date
mv=plvel(jd(2),4)               @ Mars velocity on optimal arrival date

vinfl3=absv(orbvel(0)-ev)       @ Compute launch v-infinity
c3l3=vinfl2**2                  @ Compute launch C3

vinfa3=absv(orbvel(tof)-mv)     @ Compute arrival v-infinity
c3a3=vinfa3**2                  @ Compute arrival C3

@ compute traj #4 parameters

cbodyn(jdl4,0,0)                @ Sun is central body
jd=c3min(ips,gjd4,0,2)          @ Find min arrival C3 orbit

tof=(jd(2)-jd(1))*spd           @ TOF for transfer
ev=plvel(jd(1),3)               @ Earth velocity on optimal launch date
mv=plvel(jd(2),4)               @ Mars velocity on optimal arrival date

vinfl4=absv(orbvel(0)-ev)       @ Compute launch v-infinity
c3l4=vinfl4**2                  @ Compute launch C3

vinfa4=absv(orbvel(tof)-mv)     @ Compute arrival v-infinity
c3a4=vinfa4**2                  @ Compute arrival C3
```

Appendix OT-E: Trajectory Optimization Results

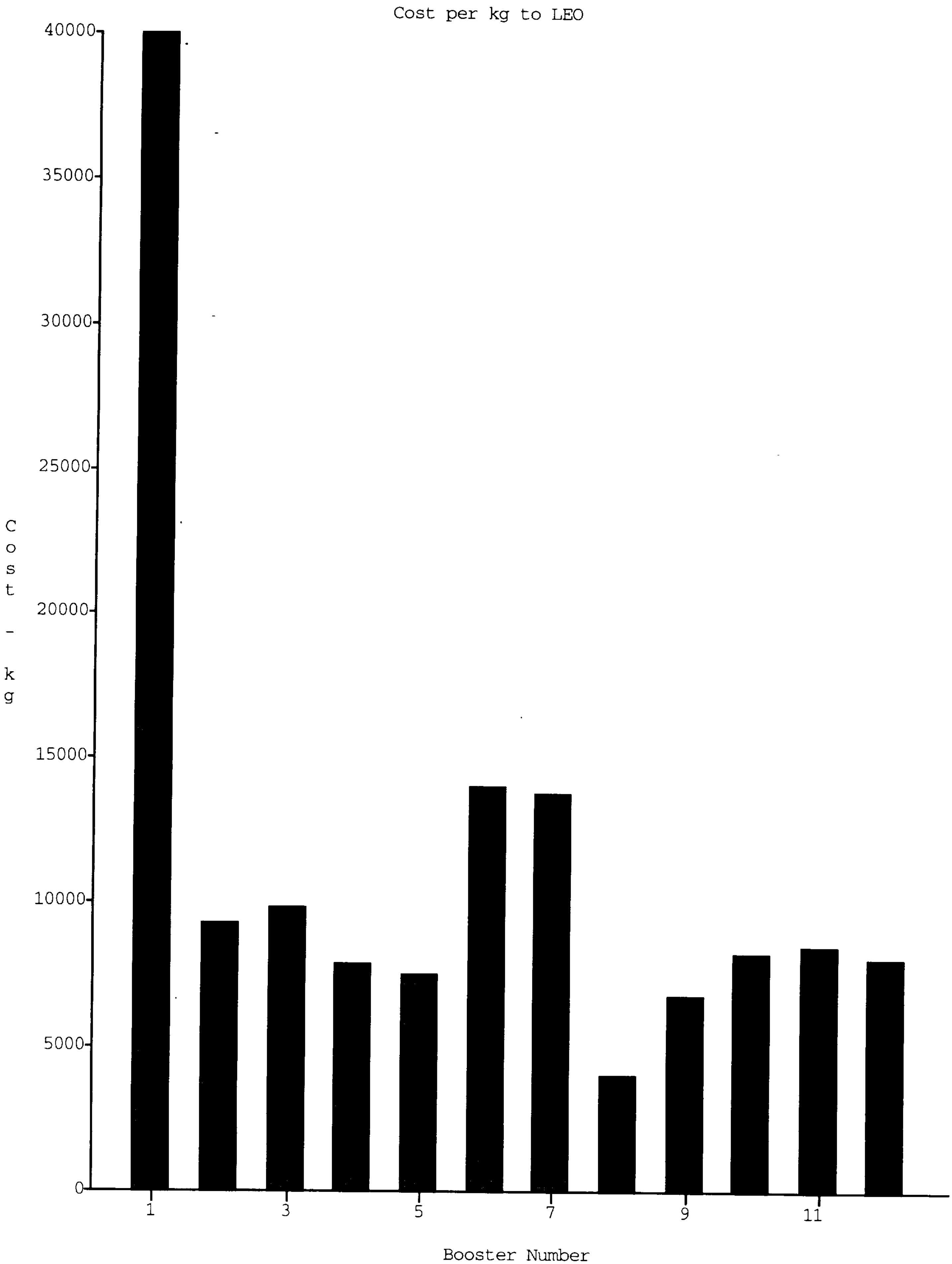
- Optimization using QUICK version 12.1 of 2/4/92
- 2002/3 Earth-Mars Ballistic Transfer Opportunity
- 4 Optimized Trajectories
  - 2 optimized for minimum launch C3, 1 Type-I and 1 Type-II
  - 2 optimized for minimum arrival C3, 1-Type I and 1 Type-II
- Minimum Launch C3 Trajectories (Trajectory-1 and Trajectory-2)
  - Trajectory-1 (Type-I)
    - Launch Date: 2452797.68 (07 June 2003)
    - Arrival Date: 2452999.34 (25 December 2003)
    - Time of Flight: 1.7423430E+07 sec [201.66 days]
    - Launch V_∞: 2.9681 km/s
    - Launch C3: 8.8095 km^2/s^2
    - Arrival V_∞: 2.7049 km/s
    - Arrival C3: 7.3163 km^2/s^2
  - Trajectory-2 (Type-II)
    - Launch Date: 2452768.98 (09 May 2003)
    - Arrival Date: 2453003.36 (29 December 2003)
    - Time of Flight: 2.0249797E+07 sec [234.37 days]
    - Launch V_∞: 3.5619 km/s
    - Launch C3: 12.6868 km^2/s^2
    - Arrival V_∞: 2.8499 km/s
    - Arrival C3: 8.1216 km^2/s^2
- Minimum Arrival C3 Trajectories (Trajectory-3 and Trajectory-4)
  - Trajectory-3 (Type-I)
    - Launch Date: 2452803.41 (12 June 2003)
    - Arrival Date: 2453005.19 (31 December 2003)
    - Time of Flight: 1.7433850E+07 sec [201.78 days]
    - Launch V_∞: 2.9960 km/s
    - Launch C3: 12.6868 km^2/s^2
    - Arrival V_∞: 2.6978 km/s
    - Arrival C3: 7.2779 km^2/s^2
  - Trajectory-4 (Type-II)
    - Launch Date: 2452770.33 (10 May 2003)

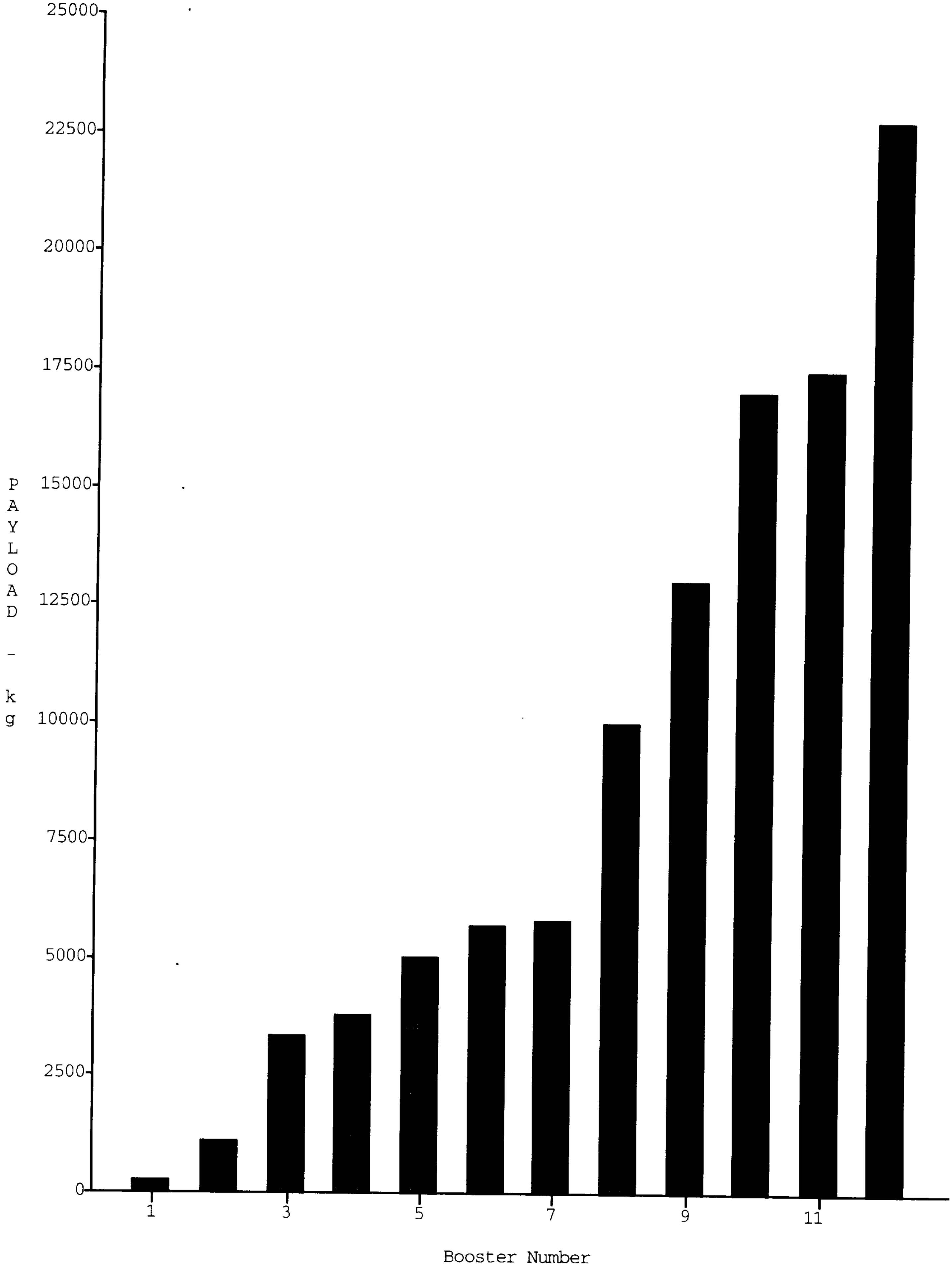
- Arrival Date: 2453006.28 (01 January 2004)
- Time of Flight: 2.0386008E+07 sec [235.95 days]
- Launch V_∞: 3.7040 km/s
- Launch C3: 13.7193 km^2/s^2
- Arrival V_∞: 2.7722 km/s
- Arrival C3: 7.6849 km^2/s^2

Appendix OT-F: Booster Cost vs. Payload Mass

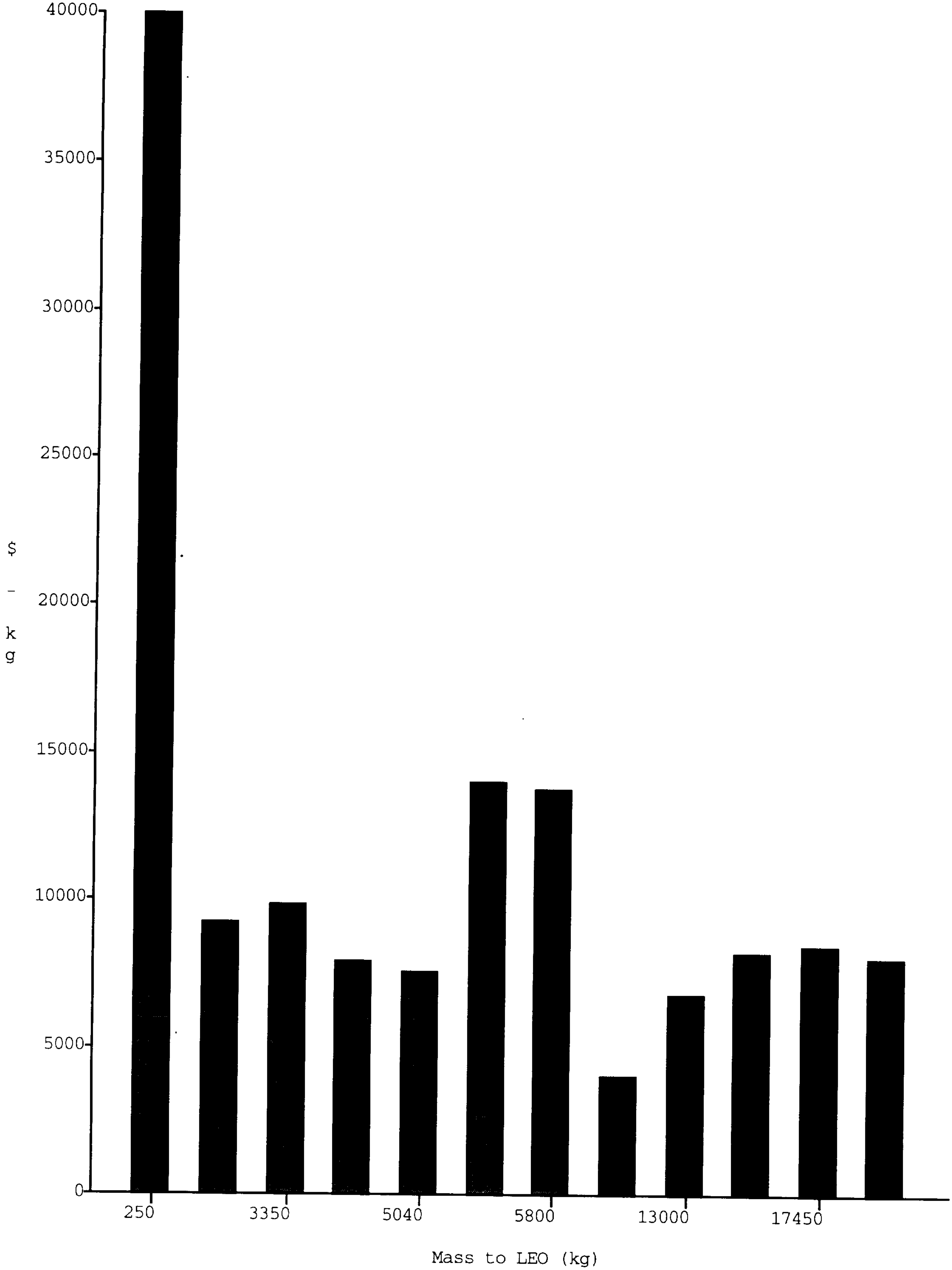
Title: Cost Data for Launch Vehicles

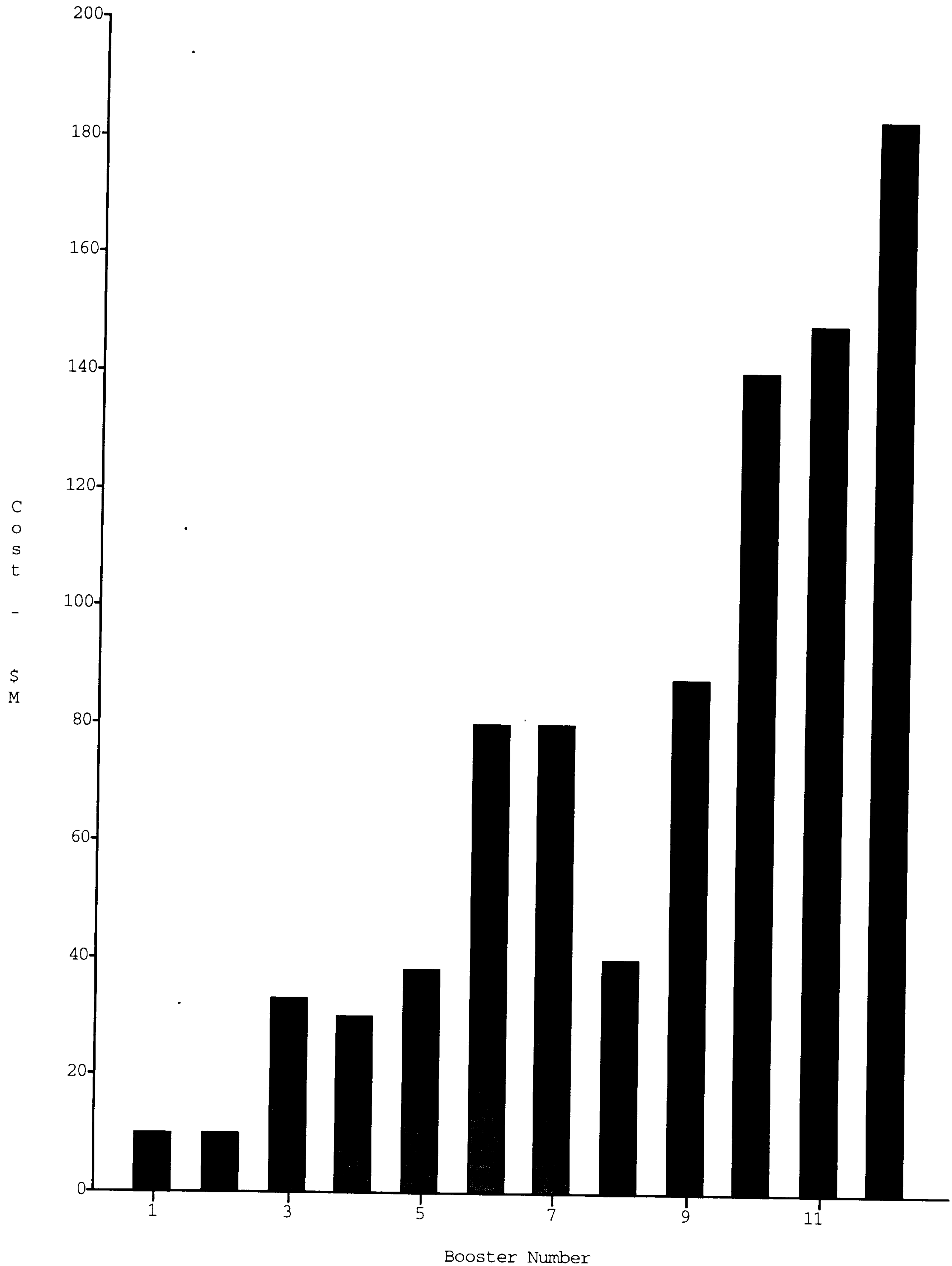
Element	Num	Booster Name	Cost ($M)	Mass to LEO (kg)	Cost/kg ($/kg)
1	1	'SCOUT	10	250	40,000
2	2	'CONESTOGA	10	1,080	9,259
3	3	'DELTA	33	3,350	9,851
4	4	'LONG_MARCH	30	3,800	7,895
5	5	'DELTA_II	38	5,040	7,540
6	6	'ARIANE_IV	80	5,700	14,035
7	7	'ATLAS_CENTAUR	80	5,800	13,793
8	8	'PROTON	40	10,000	4,000
9	9	'TITAN_III	88	13,000	6,769
10	10	'ARIANE_V	140	17,000	8,235
11	11	'TITAN_IV	148	17,450	8,481
12	12	'SHUTTLE	183	22,765	8,039


Cost per kg to LEO
40000
35000
30000
25000
20000
15000
10000
5000
0
C o s t - k g
1
3
5
7
9
11
Booster Number



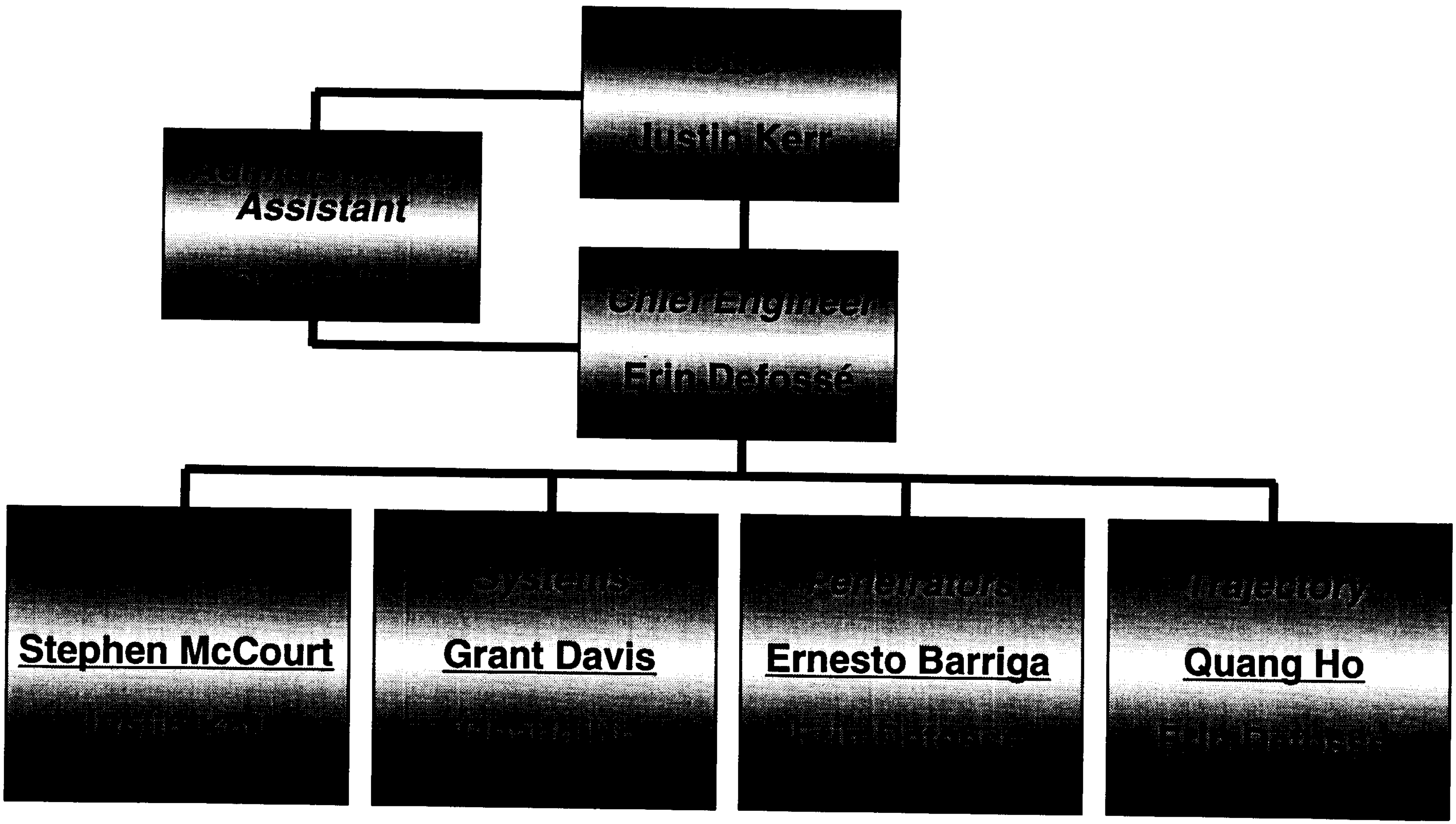
25000
22500
20000
17500
15000
12500
10000
7500
5000
2500
0
PAYLOAD - kg
1
3
5
7
9
11
Booster Number


40000
35000
30000
25000
20000
15000
10000
5000
0
$ - kg
250
3350
5040
5800
13000
17450
Mass to LEO (kg)



200
180
160
140
120
100
80
60
40
20
0
Cost - $M
1
3
5
7
9
11
Booster Number

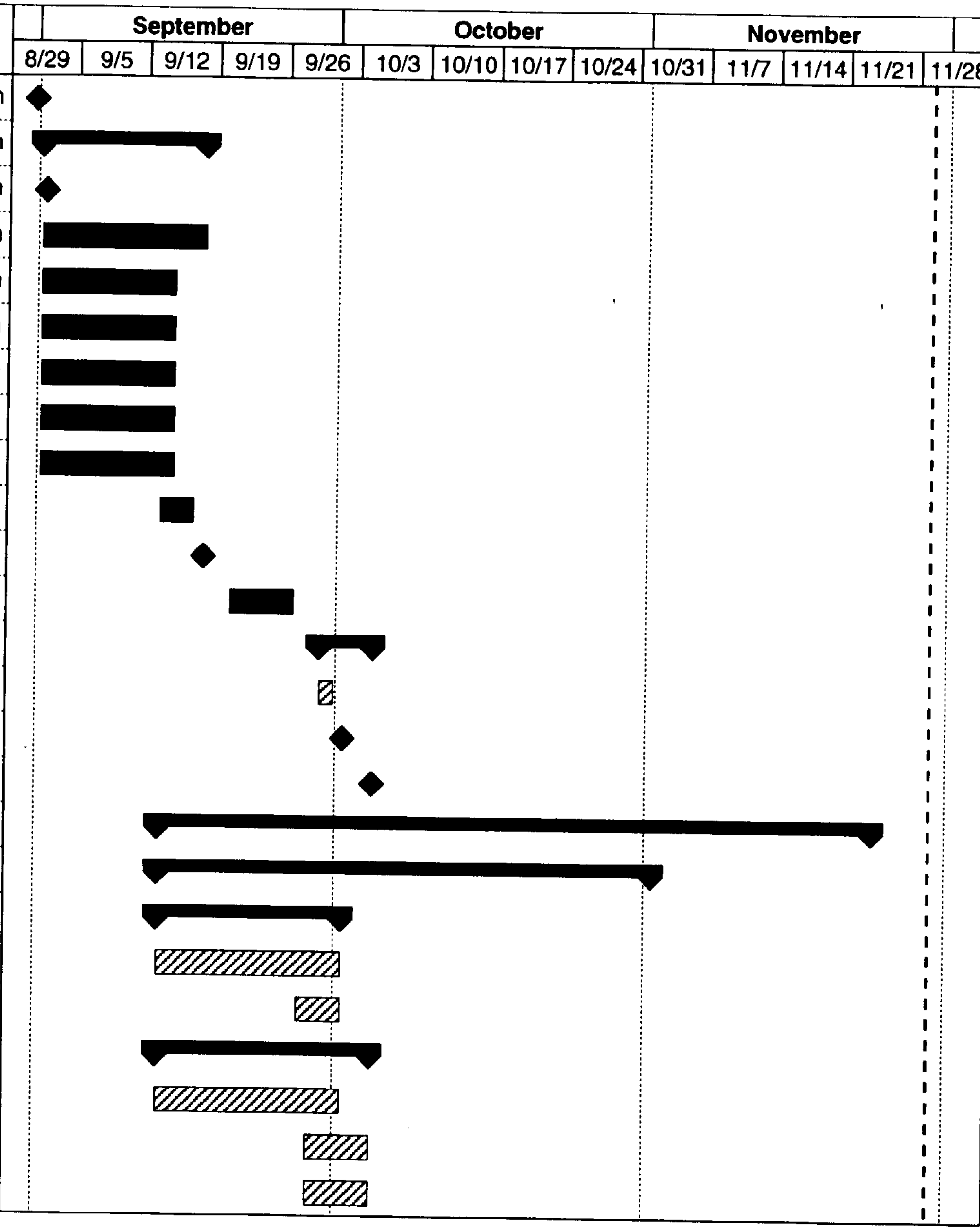
Appendix M-A: Argos Organizational Chart



Justin Kerr
Assistant
Chief Engineer
Erin Defossé
Stephen McCourt
Systems
Grant Davis
Penetrators
Ernesto Barriga
Trajectory
Quang Ho

Appendix M-B: Aeneas Project Schedule

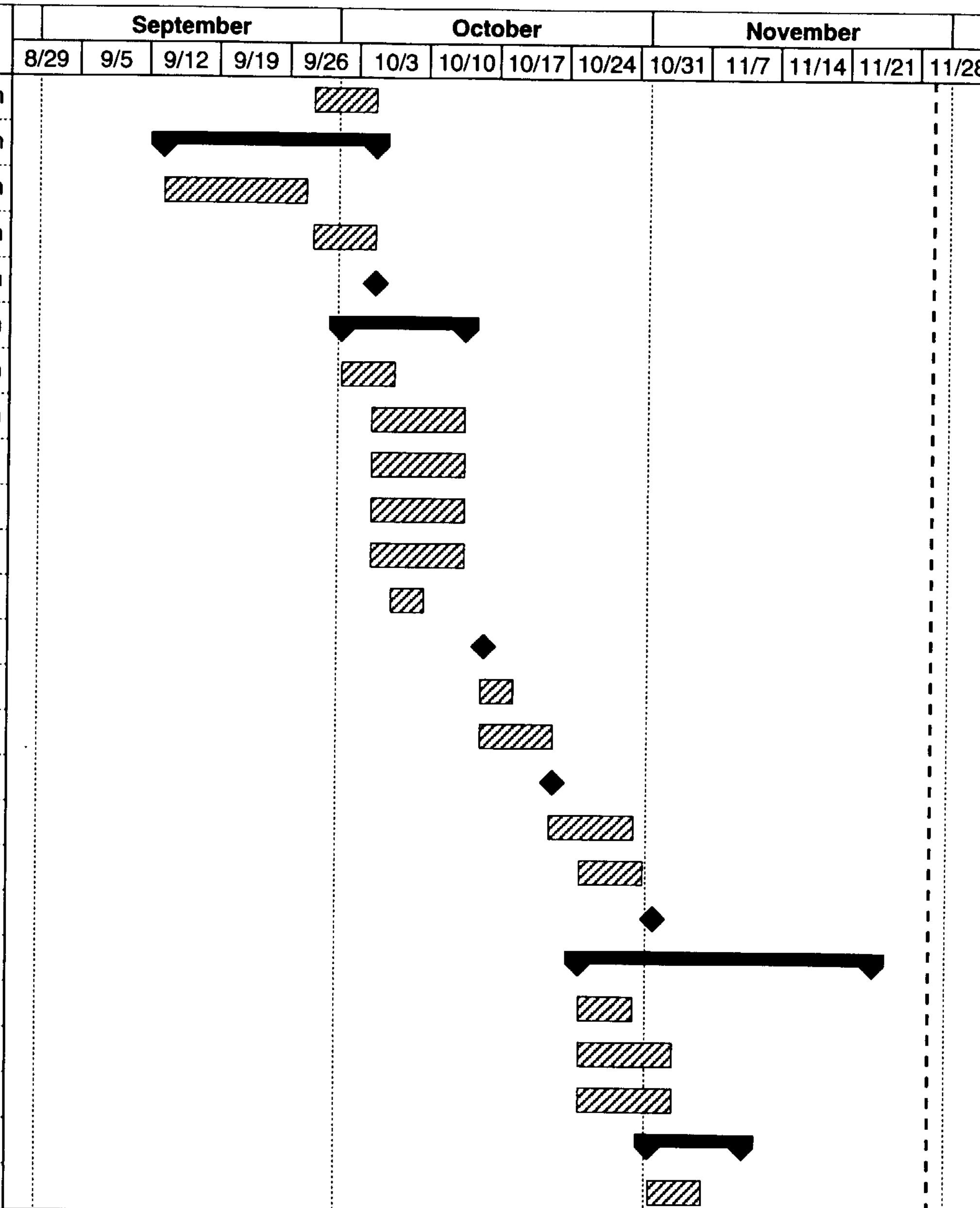
ID	Name	Duration	Scheduled Start
1	Project Start	1d	8/31/93 8:00am
2	Preliminary Activities	13d	9/1/93 8:00am
3	Team Organization	1d	9/1/93 8:00am
4	Conceptual Design	13d	9/1/93 8:00am
5	Brainstorming	10d	9/1/93 8:00am
6	Research Science Goals	10d	9/1/93 8:00am
7	Research Probe/Rover Options	10d	9/1/93 8:00am
8	Research Launch Veh. Constr.	10d	9/1/93 8:00am
9	Element Roles/Respons. Def.	10d	9/1/93 8:00am
10	CDR Preparation	4d	9/13/93 8:00am
11	CDR	1d	9/17/93 8:00am
12	Include CDR Comments	5d	9/20/93 8:00am
13	Proposal Preparation	4d	9/29/93 8:00am
14	Evaluate Requirements	2d	9/29/93 8:00am
15	Request Requirement Change	1d	10/1/93 8:00am
16	Proposal Due	1d	10/4/93 8:00am
17	Preliminary Design	52d	9/13/93 8:00am
18	PDR1	36d	9/13/93 8:00am
19	Orbit/Trajectory--Iteration1	15d	9/13/93 8:00am
20	Orbit/Trajectory Analysis	15d	9/13/93 8:00am
21	Fuel-Mass Requirements	5d	9/27/93 8:00am
22	Scientific Instr.--Iteration1	16d	9/13/93 8:00am
23	Scientific Instrument Des	15d	9/13/93 8:00am
24	SI Weight Analysis	5d	9/28/93 8:00am
25	SI Power Analysis	5d	9/28/93 8:00am



Project: Aeneas
Date: 11/29/93

Critical
Noncritical
Progress
Milestone
Summary
Rolled Up

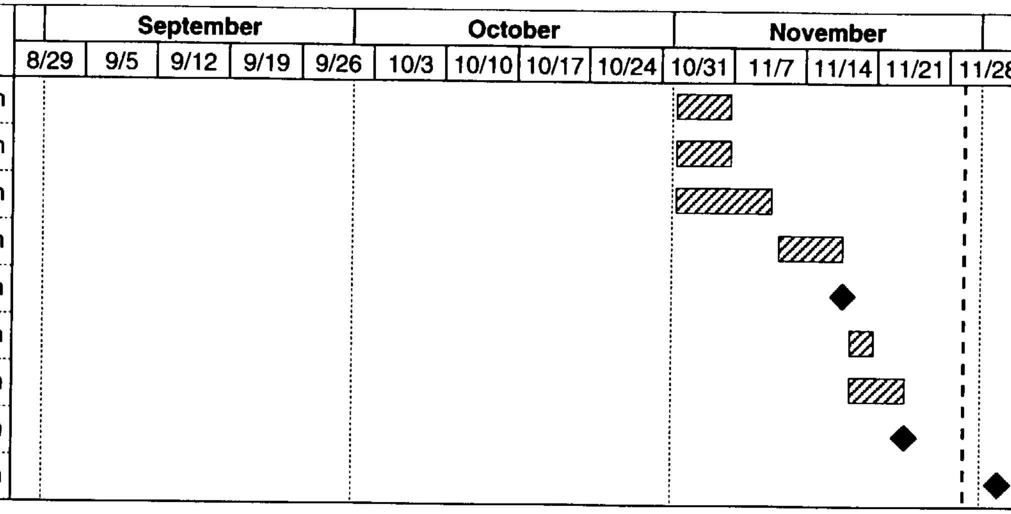
ID	Name	Duration	Scheduled Start
26	SI Communications Anal	5d	9/28/93 8:00am
27	Probe/Rover--Iteration1	16d	9/13/93 8:00am
28	Probe/Rover Description	11d	9/13/93 8:00am
29	Probe/Rover Trade Stud	5d	9/28/93 8:00am
30	Subsystem Requirements Du	1d	10/4/93 8:00am
31	Spacecraft--Iteration1	9d	10/1/93 8:00am
32	Mars Capture Orbit Sys T	4d	10/1/93 8:00am
33	GN&C Development	8d	10/4/93 8:00am
34	Communication Develop	8d	10/4/93 8:00am
35	Power System Developm	8d	10/4/93 8:00am
36	Structures Development	8d	10/4/93 8:00am
37	Launch Vehicle Trade St	3d	10/6/93 8:00am
38	Internal Review	1d	10/15/93 8:00am
39	Review Changes--Iteration 2	2d	10/15/93 8:00am
40	Prepare PDR1	6d	10/15/93 8:00am
41	PDR1 Presentation	1d	10/22/93 8:00am
42	Include PDR1 Comments	6d	10/22/93 8:00am
43	Prepare PDR1 Report	5d	10/25/93 8:00am
44	PDR1 Report Due	1d	11/1/93 8:00am
45	PDR2	22d	10/25/93 8:00am
46	Final Orbit Analysis	5d	10/25/93 8:00am
47	Finalize Subsystem Reqmts	8d	10/25/93 8:00am
48	Complete Probe/Sci. Inst Intg	8d	10/25/93 8:00am
49	Finalize Spacecraft Subsys	8d	11/1/93 8:00am
50	Complete GN&C develop	5d	11/1/93 8:00am



Project: Aeneas
Date: 11/29/93

Critical · Noncritical · Progress · Milestone · Summary · Rolled Up

ID	Name	Duration	Scheduled Start
51	Complete Comm develop	5d	11/1/93 8:00am
52	Complete Power develop	5d	11/1/93 8:00am
53	Complete Structures Ana	8d	11/1/93 8:00am
54	Prepare PDR2 Presentation	5d	11/11/93 8:00am
55	PDR 2 Presentation	1d	11/17/93 8:00am
56	Include PDR2 Comments	2d	11/18/93 8:00am
57	Prepare PDR2 Report	4d	11/18/93 8:00am
58	PDR2 Report Due	1d	11/23/93 8:00am
59	USRA/NASA Briefing	1d	12/2/93 8:00am



Project: Aeneas
Date: 11/29/93

Critical
Progress
Summary
Noncritical
Milestone
Rolled Up